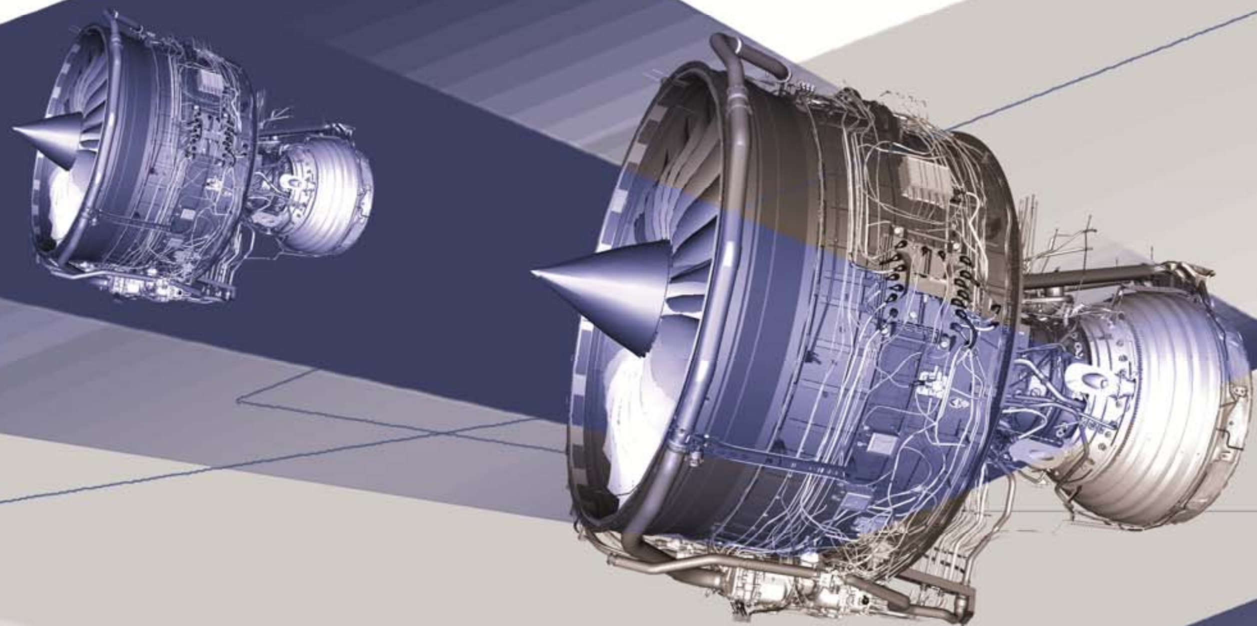
Trent XWB (Airbus A350-800/900)

Line and Base Maintenance



PREFACE

Notice to Holders

Positional Referencing.

It is to be noted that throughout these course notes any reference to a position or unit location is referred to as being viewed from the rear, unless otherwise stated. This is presuming the student is standing at the rear of the engine and/or aircraft and looking forward.

COURSE OBJECTIVES

At the end of the course, students will be able to demonstrate:

- They can describe the Trent XWB engine operation and configuration.
- They can define the Trent XWB engine systems.
- Describe; identify and locate the major Trent XWB engine Line Replaceable Units (LRUs).
- Have an ability to undertake routine Trent XWB maintenance tasks in accordance with Aircraft Documentation.
- They can explain the removal and installation procedures of some LRUs.
- They can define the deactivation procedures of certain LRUs.
- They can summarise the engine ground tests and where they may be used.
- Have an ability to define the troubleshooting process of the Trent XWB.

Rolls-Royce/Airbus Business Agreement

The business agreement introduces new business and technical interfaces. The propulsion system is divided into three work packages:

RR Responsibilities

- The bare engine and associated engine systems.
- The Engine Section Stators (ESS) anti-icing system.
- The Variable Frequency Generator Oil Cooling System.
- The technical definition of the nacelle internal aerodynamics.
- Engine Mounts.
- Fire and Overheat detection system.
- Engine Fire Extinguishing System.
- Hydraulic pipe-work.
- Engine Bleed Air System (EBAS) pipe-work and IP check valve
- Inlet anti-icing valves, controllers including pipe-work.

Airbus Responsibilities

- Variable Frequency Generators (VFG) and QAD adaptor.
- Hydraulic Pumps.
- Aircraft mounted Airborne Vibration Monitoring equipment.
- Engine Bleed Air System (EBAS) valves only (not IP check valve).
- Engine Inlet Cowl assembly (Manufacture).

UTAS Responsibilities

- Fan Cowl Assembly.
- Thrust Reverser Assembly including actuation system.
- Inlet Cowl (In service maintainability).
- Power Door Operating System (PDOS).
- Nacelle Hold Open Rods.
- Exhaust Nozzle System

ABBREVIATIONS

A	
Abs	Absolute
AC	Alternating Current
A/C	Aircraft
ACARS	Airplane Communication Addressing and Reporting System
ACMF	Aircraft Condition Monitoring Function
A/D	Analogue to Digital
ADIRS	Air Data Inertial and Reference System
AFDX	Avionics Full Duplex Ethernet
Al	Alumel (Aluminium alloy in thermocouples)
ALT	Altitude or Alternate
Amb	Ambient
AMM	Aircraft Maintenance Manual
Approx	Approximate (ly)
APU	Auxiliary Power Unit
ARINC	Aeronautical Radio Incorporated
ATA	Air Transport Association
AVM	Airborne Vibration Monitor
B	
BITE	Built In Test Equipment
BV	Bleed Valve
C	
°C	Degree Celsius
CAA	Civil Aviation Authority
CAS	Calibrated Air Speed
CDS	Controls & Data Services
C&DS	Control & Display System
CFB	Centrifugal Breather
CLB	Climb
CLR	Clear (on cockpit push button)
CLSD	Closed
CMS	Central Maintenance System
CTL	Control
D	
dB	Decibel
DC	Direct Current
DECEL	Decelerate, Deceleration
DEP	Data Entry Plug
Deg F	Degree Fahrenheit
Deg C	Degree Centigrade
DISCH	Discharge
DMC	Display Management Computer
DND	Droop Nose Device
DP	Differential Pressure
DRTO	De-Rated Take-Off
DU	Display Unit
E	
EAI	Engine Anti-Ice
EASA	European Aviation Safety Agency
EBAS	Engine Bleed Air System
EBU	Engine Build Up
ECAM	Electronic Centralised Aircraft Monitoring System
ECS	Environmental Control System
ED	Engine Display
EDP	Engine Driven Pump
EEC	Engine Electronic Controller
EECS	Engine Electronic Control System
EEPROM	Electrically Erasable Programmable Read only Memory EFIS Electronic Flight Instrument System
EGB	External Gearbox

EGT	Exhaust Gas Temperature
EIF	Engine Interface Function
EIS	Electronic Instrument System
Ems	Engine Monitoring System
EMU	Engine Monitor Unit
ENG	Engine
EPDS	Electronic Power Distribution System
ETRAC	Electric Thrust Actuation System Controller
ETRAS	Electrical Thrust Reverser Actuation System
E/WD	Engine / Warning Display
ESS	Engine Section Stators
ESSAIV	Engine Section Stator – Anti-Ice Valve
ESN	Engine Serial Number
ETOPS	Extended Range Twin Operational Performance Standards
EXT PXR	External Power
F	
°F	Degree Fahrenheit
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Control
FAV	Fan Air Valve
FBH	Front Bearing Housing
FCD	Fan Cowl Door
FCSB	Fan Cowl Door Support Beam
FCU	Flight Control Unit
FDR	Flight Data Recorder
FFTX	Fuel Flow Transmitter
FLT	Flight
FLEX	Flexible Take-Off Rating
FMS	Flight Management System
FMV	Fuel Metering Valve
FOGV	Fan Outlet Guide Vanes
FOHE	Fuel Oil Heat Exchanger
FPF	Fire Protection Function
FSN	Fuel Spray Nozzle
FREQ	Frequently
FWC	Flight Warning Computer
FWD	Forward
FWS	Flight Warning System
G	
GBX	Gearbox
gpm	Gallons per Minute
GND	Ground
GSE	Ground Support Equipment
H	
HCU	Hydraulic Control Unit
HMU	Hydro-Mechanical Unit
HOR	Hold Open Rod
HP	High Pressure
HPC	High Pressure Compressor
HPV	High Pressure Valve
HPT	HP Turbine
Hz	Hertz
I	
I/O	Input / Output
IFS	Inner Fixed Structure
IFSD	In Flight Shut Down
IGB	Intermediate Gear Box
IGN	Ignition
IGV	Inlet Guide Vane
IP	Intermediate Pressure
IPBV	IP Bleed Valve
IPC	Illustrated Parts Catalogue
IPC	Intermediate Pressure Compressor
IPT	Intermediate Pressure Turbine
ISA	International Standard Atmosphere
IV	Isolation Valve

J

JB	Junction Box
JCT	Junction

K

°K	Degree Kelvin
K	Kilo
Kg	Kilogram
KGPH	Kilogram per Hour
KT	Knot
KV	Kilo Volt
KVA	Kilo Volt Ampere

L

L	Left
ELE	Elliptical Leading Edge
lb(s)	Pound(s) (weight)
lbf	Pounds Force
lbs / hr	Pounds per Hour
LE	Leading Edge
L/G	Landing Gear
LH	Left Hand
LP	Low Pressure
LPC	Low Pressure Compressor
LPT	Low Pressure Turbine
LPTCCV	LP Turbine Case Cooling Valve
LPTOS	Low Pressure Turbine Overspeed
LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transducer

M

MAINT	Maintenance
MAX	Maximum
MCD	Magnetic Chip Detector
MCDU	Multi-Purpose Control Display Unit
MCT	Maximum Continuous Thrust
MDU	Manual Drive Unit
MEL	Minimum Equipment List
MMEL	Master Minimum Equipment List
MHz	Megahertz
MIN	Minimum
mm	Millimetres
MN	Mach Number
MOD	Modification
ms	Millisecond
MTBF	Mean Time Between Failure
MTBR	Mean Time Between Removals
MTO	Maximum Take-Off
mV	Millivolts

N

N	Rotational Speed
N1	Low Pressure Assembly Speed
N2	Intermediate Pressure Assembly Speed
N3	High Pressure Assembly Speed
N3 dot	Rate of Change of N3
ND	Navigation Display
NGV	Nozzle Guide Vane
NH	HP Shaft Speed
NI	IP Shaft Speed
NL	NL Shaft Speed
NLC	LP Compressor Speed
NLT	LP Turbine Shaft Speed
NRV	Non Return Valve
NVM	Non Volatile Memory

O

ODS	Oil Debris Sensor
ODSC	Oil Debris Signal Conditioner

OGV	Outlet Guide Vane
OPS	Overspeed Protection System
OPV	Over Pressure Valve
OVHT	Over Heat
OVSPD	Overspeed
P	
P0	Ambient Pressure
Ps160	Fan Exit Pressure
P20	Engine Intake Pressure
P25	IP Compressor Exit Pressure
P30	HP Compressor Delivery Pressure
Ps42	IPT Inlet Pressure
Ps44	IPT Exit Pressure
Pamb	Ambient Pressure
P/B	Pushbutton
PDOS	Power Door Opening System
PFD	Primary Flight Display
PMA	Permanent Magnet Alternator
Poil	Oil Pressure
POSN	Position
PPH	Pounds Per Hour
PRESS	Pressure
Prox	Proximity Sensor
PRSOV	Pressure Regulating Shut-Off Valve
PRV	Pressure Regulating Valve
PS	Pressure Switch
PSI	Pounds per Square Inch
PSIA	Pounds per Square Inch Absolute
PSID	Pounds per Square Inch (Differential)
PSIG	Pounds per Square Inch Gauge
PWR	Power
Q	
QAD	Quick Attach Detach
QEC	Quick Engine Change
Qoil	Oil Quantity
QTX	Quantity Transmitter
QTY	Quantity
R	
R	Right
RAT	Ram Air Turbine
RAD ALT	Radio Altitude
REF	Reference
REV	Reverser
RTD	Resistive Temperature Device
RVDT	Rotary Variable Differential Transducer
S	
SC	Signal Conditioner
S/C	Short Circuit
S/D	Shut Down
SAT	Static Air Valve
SB	Service Bulletin
SCV	Start Control Valve
SCU	Signal Conditioning Unit
SD	System Display
sec	Second
SED	Secondary Engine Display
SER NO	Serial Number
SFC	Specific Fuel Consumption
SLS	Sea Level Static
Sol	Solenoid
SOV	Shut-Off Valve
SW	Switch
SYS	System
T	
T0	Ambient Air Temp
T20	Engine Intake Temperature
T24	IP Compressor Inlet Temperature (Synthesised)
T25	IP Compressor Exit Temperature (Measured)

T30	HP Compressor Exit Temperature
T50	LP Turbine Exit Temperature
TAI	Thermal Anti-Ice
TAT	Total Air Temperature
TBD	To Be Determined / Decided
TBH	Tail Bearing Housing
TC	Thermocouple
TCAF	Turbine Cooling Air Front
TCAR	Turbine Cooling Air Rear
TCC	Turbine Case Cooling
TCCV	Turbine Case Cooling Valve
TCM	Thrust Control Malfunction
TDC	Top Dead Centre
TE	Trailing Edge
TEMP	Temperature
T fuel	Fuel Temperature
TGB	Transition Gear Box
TGT	Turbine Gas Temperature
THR	Thrust
TLA	Throttle Lever Angle
TOGA	Take-Off/Go Around
Toil	Oil Temperature
TOS	Turbine Overspeed
TRA	Throttle Resolver Angle
TRAS	Thrust Reverser Actuation System
TRU	Transformer Rectifier Unit
TSN	Time Since New
TSO	Time Since Overhaul
U	
U/S	Unserviceable
V	
V	Volts
VAC	Volts Alternating Current
VCAS	Calibrated Airspeed
VFG	Variable Frequency Generator
VIGV	Variable Inlet Guide Vane
VSV	Variable Stator Vane
VSVA	Variable Stator Vane Actuator
W	
WOW	Weight on Wheels

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Section 1 - Introduction

Trent XWB for the Airbus A350 Aircraft

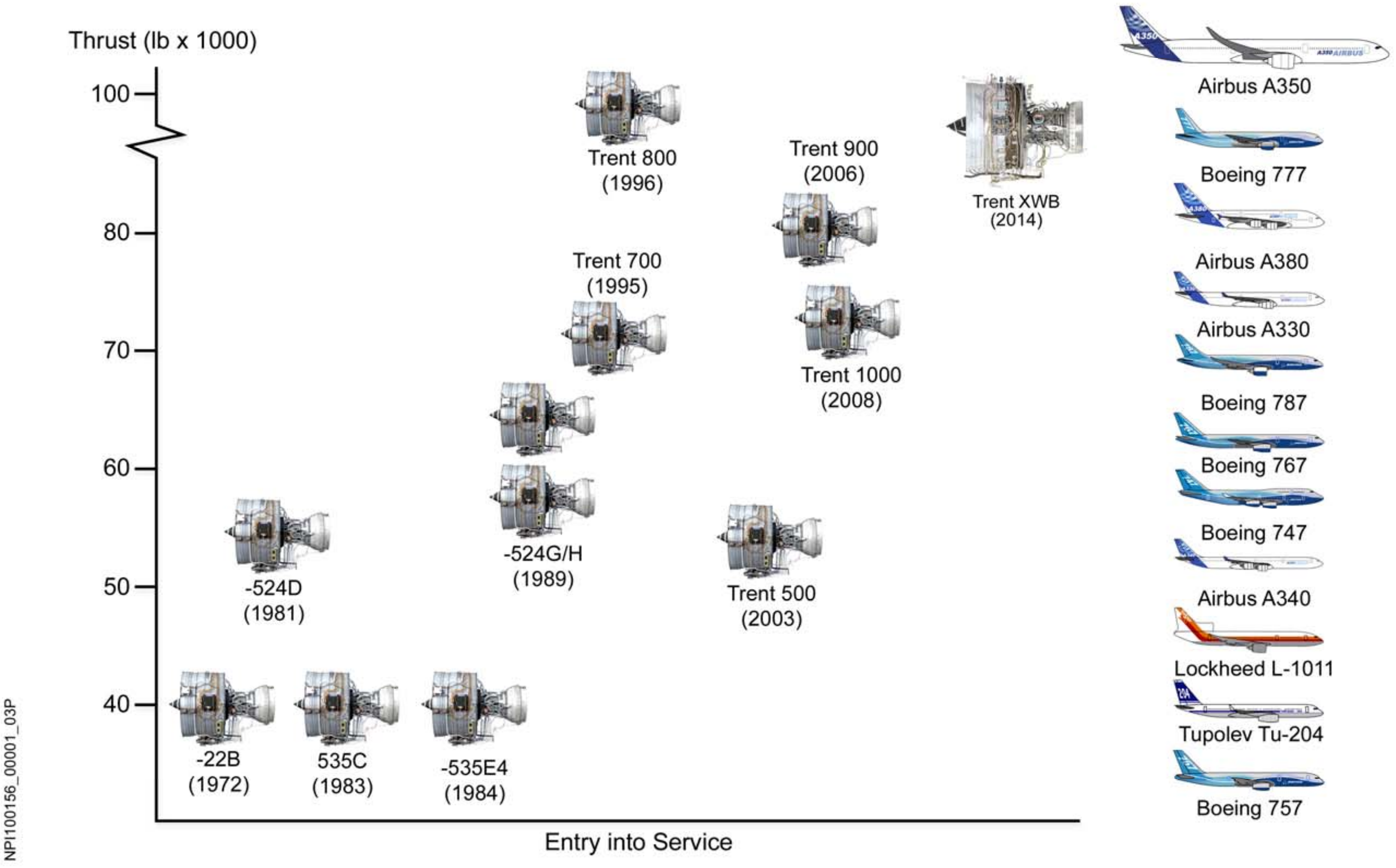
Rolls-Royce has developed the Trent family of engines to meet the market demand for medium-to-large, long-range commercial aircraft.

The existing Trent family, which utilises proven advanced technology, has provided many airlines with a low-risk solution to reliable engine power, accumulating more than 35 million + hours since the first of the family, the Trent 700, entered service in 1995.

The Trent XWB is the sixth generation of the Trent family and is the launch engine for the Airbus A350 aircraft. It exploits unique new technologies to deliver the best ever performance at the lowest life cycle cost with minimal environmental impact.

Maturity at Entry into Service (EIS) is ensured by the use of high technology components developed with the benefit of Trent family experience. The unique Rolls-Royce three-concentric shaft configuration combined with a very high bypass ratio and enhanced component efficiencies contribute to significantly improved fuel consumption.

The Trent XWB has been designed to provide power for the entire Airbus A350 family with the best economics at the lowest risk.



THE RB211 & TRENT FAMILY

Propulsion System Outline

The Airbus A350 is powered by two propulsion systems attached to pylons on the underside of each wing.

The Rolls-Royce Trent XWB engine is described as a three-shaft high bypass ratio turbofan engine and is the first Trent engine to incorporate one piece disc and blade combination (Blisk) into the compressor design.

The shafts are identified as the Low Pressure (LP) system, Intermediate Pressure (IP) system and High Pressure (HP) system.

The LP system consists of a single stage compressor (Fan) assembly driven by a six-stage turbine.

The IP system consists of an eight-stage axial flow compressor driven by a two-stage turbine.

The HP system consists of a six-stage axial flow compressor driven by a single stage turbine.

Air enters the engine through the air inlet cowl and is compressed by the LP Compressor (Fan). The exiting air is then split into two main flows, bypass and core.

The bypass air, which has a greater mass flow and produces the majority of the overall thrust of the engine, passes through the raked Fan Outlet Guide Vanes (FOGV) into the bypass duct before exiting the outside of the cold exhaust nozzle.

The IP and HP Compressors compress the core flow further before entering the combustion chamber. Fuel is added to the compressed air and the resulting mixture is ignited. The gases expand through the turbines where the energy is extracted to drive the compressors before being exhausted through the hot side of the centre body of the exhaust nozzle.

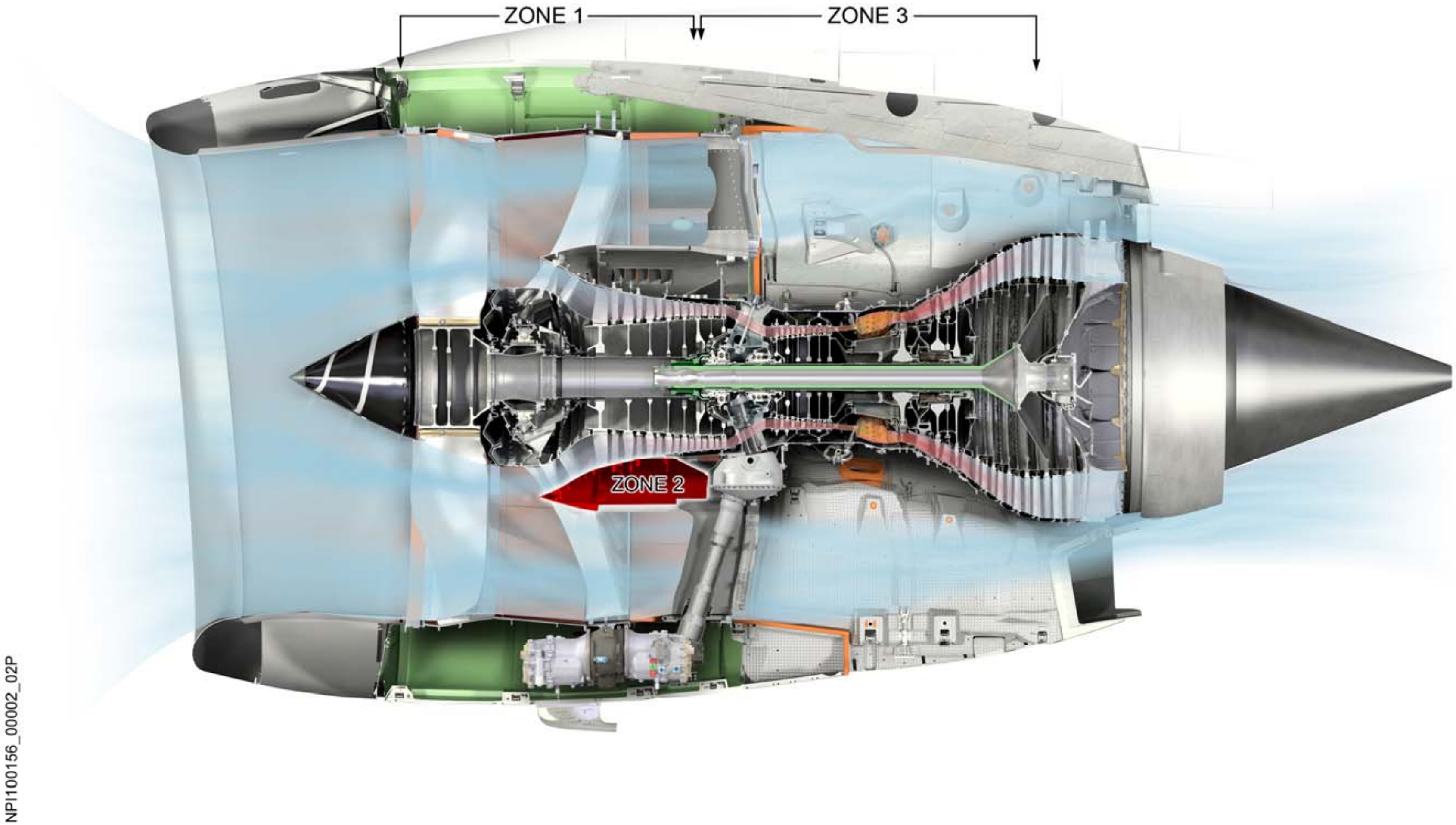
The combustion system is of annular construction incorporating 20 fuel spray nozzles through which fuel is supplied from the aircraft fuel system, in response to engine throttle setting and aircraft operating conditions.

The External Gearbox (EGB) is mounted at the bottom of the fan case. A driveshaft connected to the HP System provides drive for the following accessories:

- Combined LP and HP Fuel Pump Assembly
- Engine Oil Pump Assembly
- Centrifugal Breather
- Two Hydraulic Pumps
- Permanent Magnet Alternator (PMA)
- Two Variable Frequency Generators (VFG)
- Hydro-Mechanical Unit (HMU) only mounted on the gearbox (none driven)

For ventilation and fire protection purposes the engine is divided into three sealed compartments. These compartments are known as Fire Zones and can be described as:

- Zone 1 – The area under the fan cowl doors
- Zone 2 – The area under the Gas Generator fairings
- Zone 3 – The area under the thrust reverser



PROPULSION SYSTEM OUTLINE

Trent XWB Leading Particulars

	Low Pressure System (N1 Indication)	Intermediate Pressure System (N2 Indication)	High Pressure System (N3 Indication)
Compressor Stages	One	Eight	Six
Turbine Stages	Six	Two	One
Direction of Rotation Looking from Rear	Counter-clockwise	Counter-clockwise	Clockwise

Overall Pressure Ratio	Between 40:1 and 48:1 Dependant On Engine Rating	Note: International Standard Atmosphere (ISA) / Sea Level Static (SLS) The Flat Rating (Kink Point) is, ISA + 15°C
By-pass Ratio	9:1 Variable	
Dry Weight	16043 lbs (7277 Kg)	
Maximum Vibration (N1)	5 Units in / secs	

Aircraft Variant	A-350-900		A-350-1000	
XWB Engine Variant	75S	84	97	
Maximum Continuous lbs. Thrust		71400	83100	
Maximum Take-off Thrust lbs.		84200	97000	

Shaft Rotation Speeds	Ground Idle	100%	Ground Idle	100%		
LP (N1) RPM		2545		2714		
IP (N2) RPM		8023		8155		
HP (N3) RPM		12005		12130		

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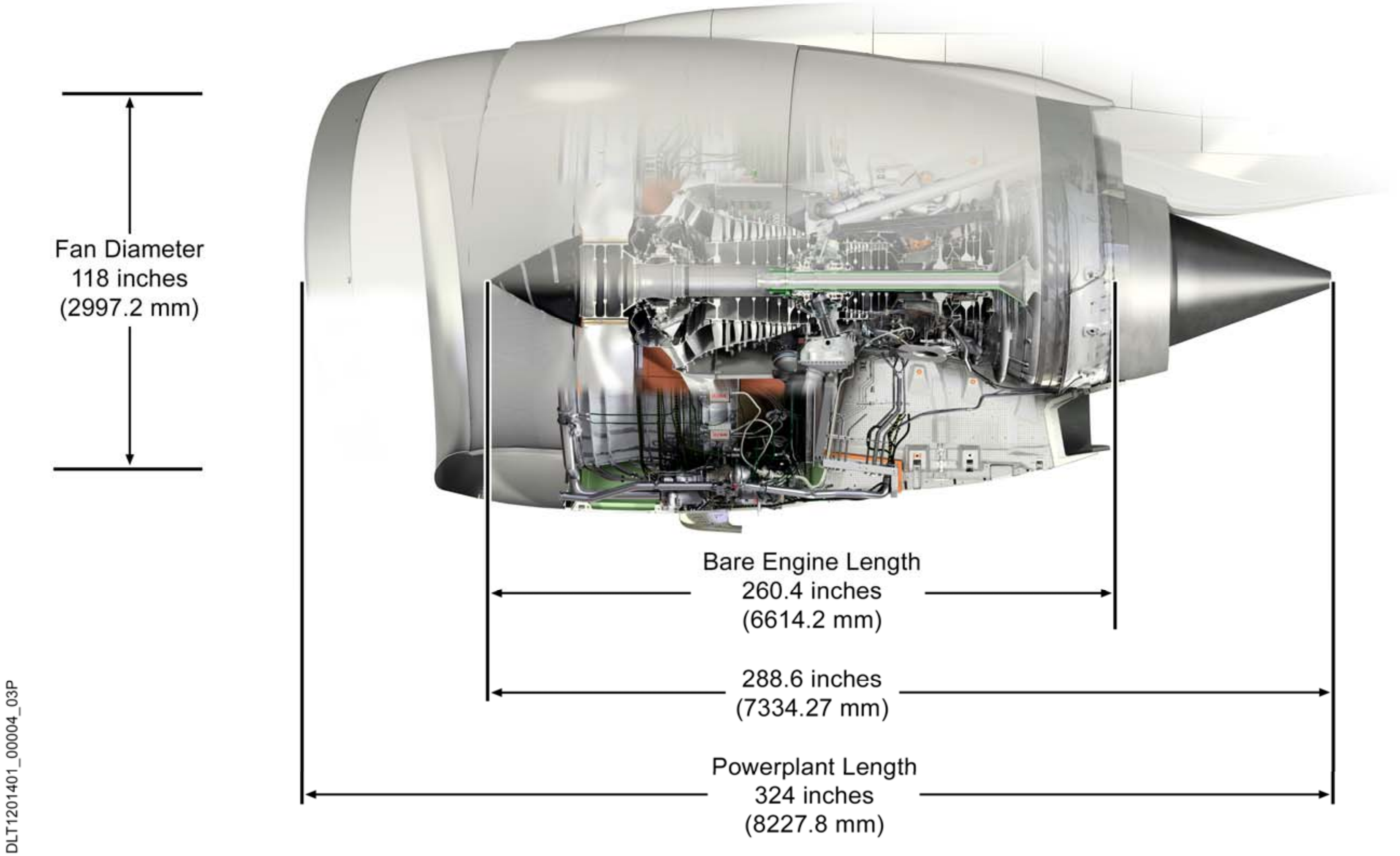
LEADING PARTICULARS



THE TRENT XWB ENGINE

Safety Notice

When the engines are operating the intake and exhaust areas can be extremely dangerous and the following warnings and cautions should be adhered to, in order to prevent serious injury to persons or damage to equipment.

WARNING

KEEP ALL PERSONS OUT OF THE DANGER AREAS DURING ENGINE OPERATION.

WARNING

CLEAN THE RAMP IF THERE IS SNOW, ICE, WATER, OIL OR OTHER CONTAMINATION OR MOVE THE AIRCRAFT TO A LOCATION THAT IS CLEAN.

WARNING

MAKE SURE THAT ALL PERSONS ARE SAFE BEFORE YOU START THE ENGINE.

WARNING

MAKE SURE THE PERSONS IN THE COCKPIT CAN SPEAK TO ALL PERSONS NEAR THE DANGER AREA DURING ENGINE OPERATION.

WARNING

OBEY ALL OF THE GROUND SAFETY PRECAUTIONS FOR THE ENGINES.

WARNING

THE ENGINES CAN PULL PERSONS OR UNWANTED MATERIALS INTO THEM AND CAUSE SERIOUS INJURIES OR DAMAGE TO EQUIPMENT.

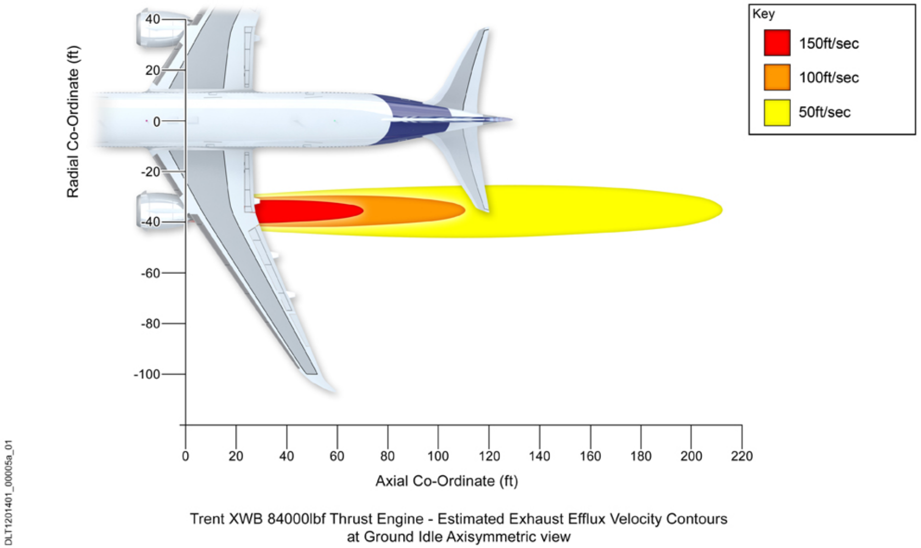
CAUTION

BE AWARE THAT AT GROUND IDLE THE FAN ROTATION SPEED AND ENGINE NOISE IS LOW, THIS CAN GIVE THE IMPRESSION THAT ENGINE IS JUST WIND MILLING AND NOT ACTUALLY UNDER POWER.

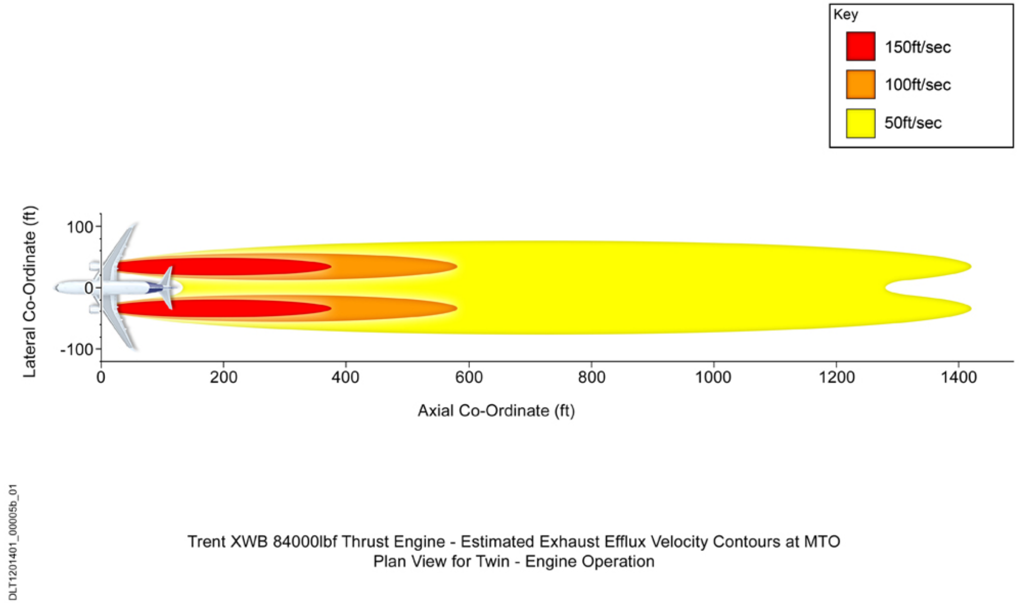
Note

The danger from compressor suction and exhaust blast increases greatly as power is increased from idle to take off settings.

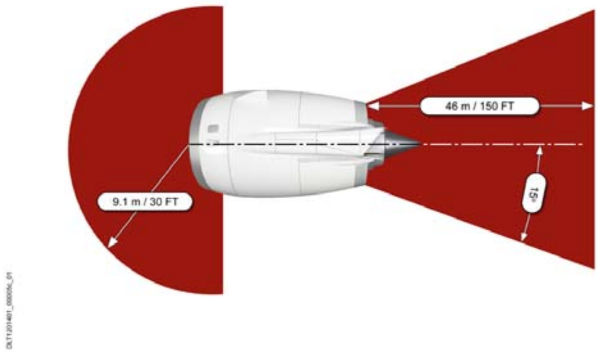
Intake Danger Area Idle	30 feet
Intake Danger Area Take Off	50 feet
Exhaust Danger Area Idle	220 feet
Exhaust Danger Area Take Off	1400 feet



Trent XWB 84000lbf Thrust Engine - Estimated Exhaust Efflux Velocity Contours
at Ground Idle Axisymmetric view



Trent XWB 84000lbf Thrust Engine - Estimated Exhaust Efflux Velocity Contours at MTO
Plan View for Twin - Engine Operation



DANGER AREAS DURING ENGINE OPERATION

Relative Wind Direction during Engine Running

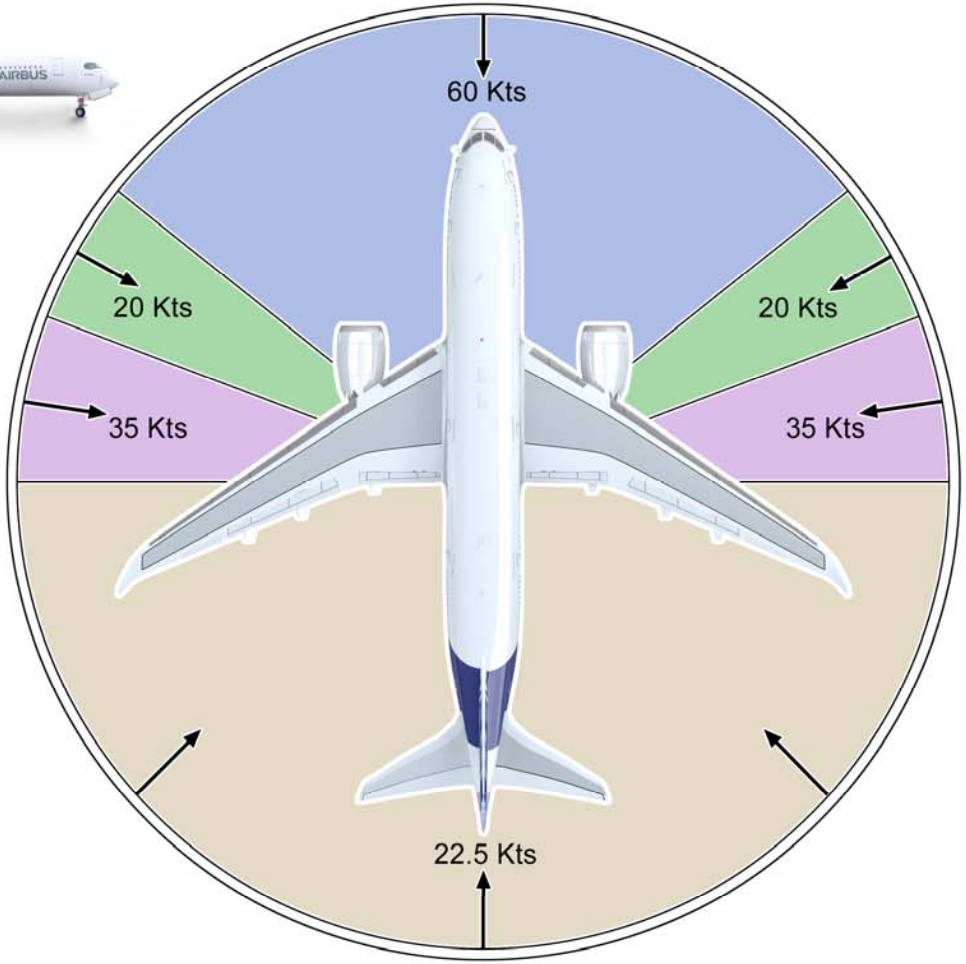
Introduction

For the engine to operate correctly the aircraft should be positioned as to prevent the possibility of the engine surging.

Persons operating the aircraft on the ground must ensure they observe the relative wind direction and velocity limits. Below is a brief summary of the relevant subtask. The Aircraft Maintenance Procedures (AMP) should be referred to in its entirety before engine operation.

Safe Operation of Engine during EGR

- Refer to the Engine Operation Limits in the AMP. Run the engine to the minimum power for the required task.
- If throttle movements are required move them slowly to prevent quick changes in engine temperature. Fast movement will ultimately decrease the life of the engine.
- Ensure the wind direction and velocity limits are observed and are in Operating limits.
- Turbulence, gusting and crosswind conditions can cause Thrust (THR), Exhaust Gas Temperature (EGT) and Revolutions per Minute (RPM) indications on the Electronic Centralised Aircraft Monitoring (ECAM) screen to be unstable at middle power and above. A variation in the tone of the engine inlet may also be heard. This will require immediate action by the operator to prevent operational exceedance. It is therefore not permitted to run in these conditions.
- A surge during engine ground running can be identified by an increase in EGT and a sudden noise from the engine. If the surge is large enough flames may be seen forward of the engine inlet. In this case the relevant steps should be taken in accordance with the Engine Operating Manual and the AMP. The Rolls-Royce Field Service Representative should also be informed.
- If the engine is operated on the ground in certain ambient conditions static electricity may be seen to discharge from the LP Compressor Nose Cone. This will not cause damage to the engine.
- Only carry out engine ground runs if the local weather conditions permit, e.g. icing conditions or side wind conditions allow, in accordance with the Engine Operating Manual and the AMP.



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TRENT XWB GROUND WIND ENVELOPE

Definition of WARNING, CAUTION and Note

The safety notices associated with the following headings should be given special attention when they appear in maintenance, operating and emergency procedures.

<u>WARNING</u>

CONCERNS AN OPERATING PROCEDURE OR PRACTISE THAT, IF NOT STRICTLY OBSERVED, CAN RESULT IN INJURY PERSONNEL OR LOSS OF LIFE

<u>CAUTION</u>

CONCERNS AN OPERATING PROCEDURE OR PRACTISE THAT, IF NOT STRICTLY OBSERVED, CAN RESULT IN DAMAGE TO OR DESTRUCTION OF EQUIPMENT

Note

Concerns an operating procedure or condition that needs highlighting.

End of Introduction Section

Section 2 – Nacelle

Section 2 - Nacelle

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance procedural level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the nacelle as fitted to the Trent XWB engine / A350.
- Locate and identify the major assemblies that form the nacelle of the Trent XWB engine / A350.
- Describe the purpose and operation of the major assemblies that form the nacelle of the Trent XWB engine / A350.
- State the WARNINGS & CAUTIONS associated with the nacelle as fitted to the Trent XWB engine / A350.
- Describe how the Trent XWB nacelle interfaces with other engine and aircraft systems.

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Nacelle Major Assemblies

Location

The major assemblies of the Nacelle are located externally around the engine.

Purpose

The purpose of the Nacelle major assemblies is to provide a structural and streamlined enclosure that fits around the engine and interfaces with the aircraft structure.

Description

The powerplant major assemblies are listed below, when combined form the engine nacelle.

- Air inlet cowl.
- Left and right fan cowl doors.
- Thrust reverser assembly.
- Hot gas exhausts system.

Air Inlet Cowl

The air inlet cowl is attached to the front flange of the LP compressor case; it is required to ensure smooth airflow into the fan inlet despite air approaching the inlet from directions other than straight ahead. The inlet cowl also helps reduce drag across the nacelle.

Fan Cowl Doors

The fan cowl doors cover the LP compressor case to provide an aerodynamic surface between the air inlet cowl and the thrust reverser. They are attached to the engine pylon and can be independently powered open (and closed) to provide access to the LP Compressor case for maintenance.

Thrust Reverser Assembly

The thrust reverser assembly consists of two halves called C-ducts. They are attached to the pylon and when operated on landing by the flight crew the outer translating sleeves move rearwards to re-direct the air in a forward direction.

To access the core components of the engine for maintenance, the C-ducts can be independently powered open (and closed).

Exhaust Nozzle System

This is made up of three sections that are attached to the rear of the engine.

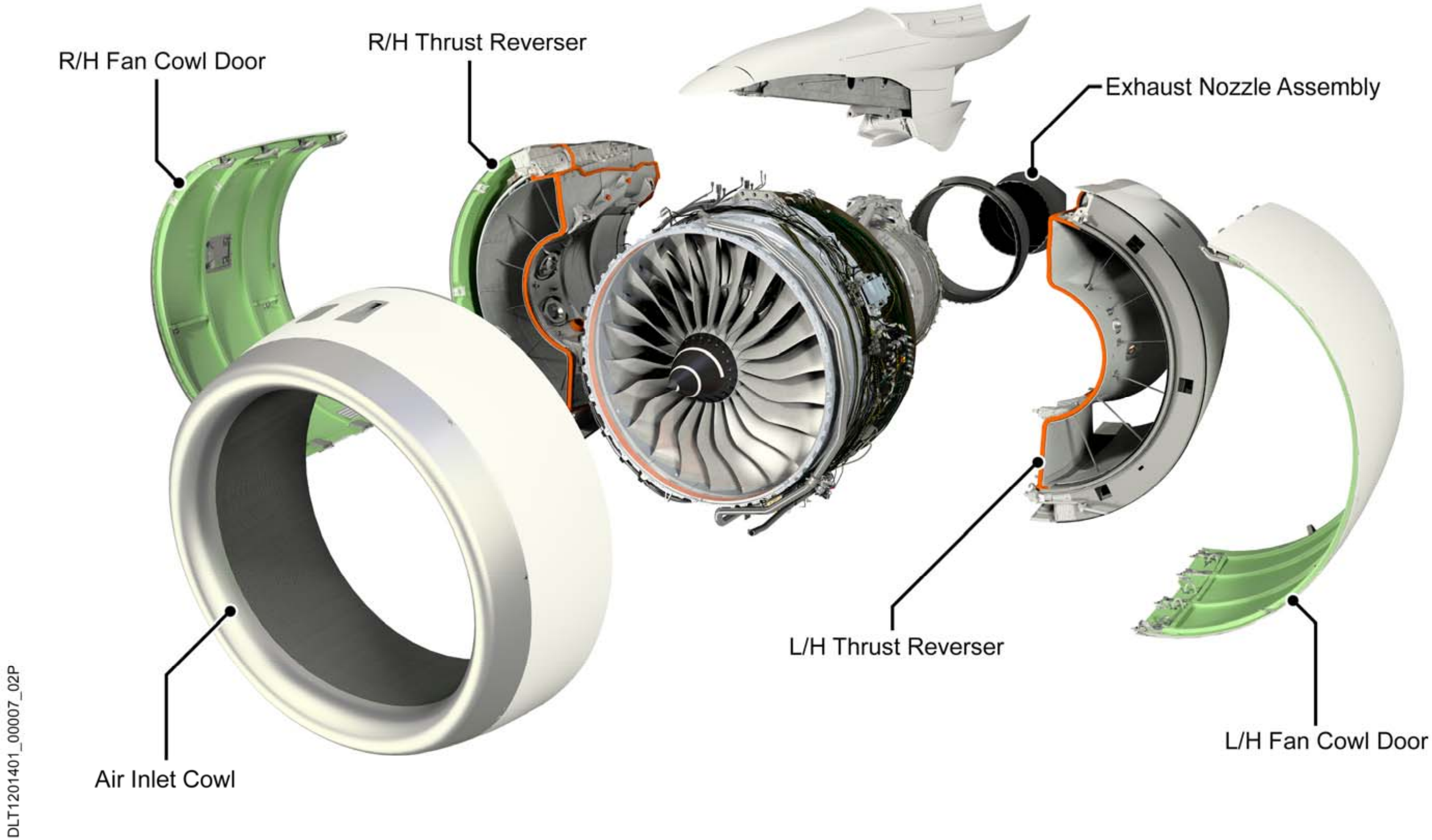
The outer exhaust nozzle

The forward centre body exhaust nozzle

The aft centre body exhaust nozzle

Note:

The C-ducts cannot be opened for maintenance access unless the fan cowl doors have been opened first.



POWERPLANT MAJOR ASSEMBLIES

Air Inlet Cowl

Location

The inlet cowl is bolted to the forward face of the engine fan case on the A1 flange with 52 nuts & bolts & spacers & 3 alignment pins.

Purpose

The inlet cowl provides smooth airflow to the engine fan and core sections and noise attenuation that reduces engine generated noise.

Description

The assembly consists of an inlet lip skin, forward bulkhead, outer barrel, acoustic single piece inner barrel, aft bulkhead, aft flange and lip skin de-icing system.

• The inlet cowl protects the engine from ice ingestion by using a thermal anti-icing system

• T20 (inlet temperature) probe is installed at 1 o’clock position

• It accommodates the Power Door Operating System (PDOS) switches

• Nacelle Anti-Ice duct (NAI) with outer shroud and outer barrel vent opening for exiting of anti-ice air exhaust

• It has 4 hoist points for Ground Support Equipment (GSE) sling

• It is a Line Replaceable Unit (LRU)

Aft Bulkhead

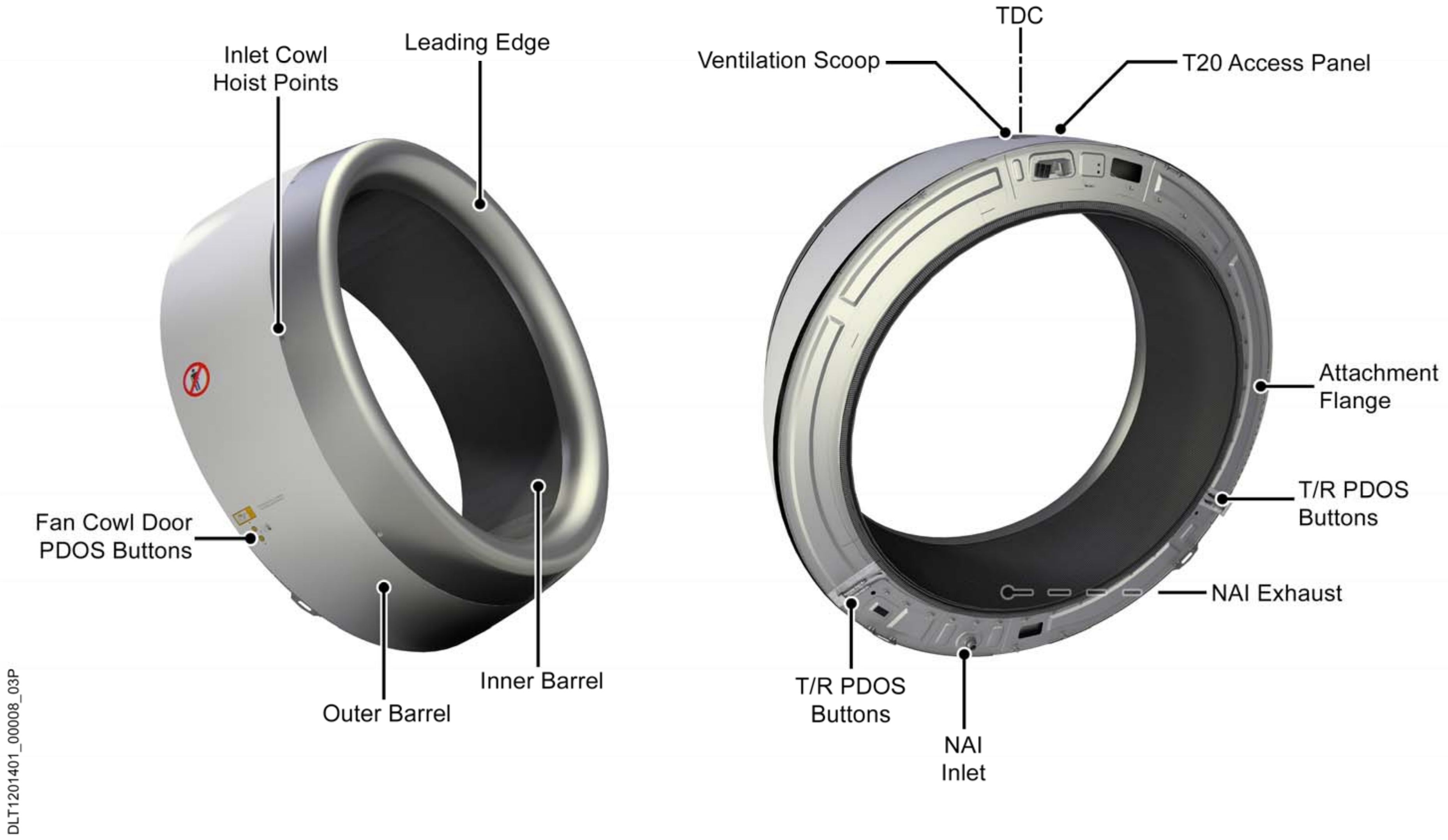
The Aft Bulkhead consists of six inner panel segments and six outer panel segments. The six inner panel segments are made of metal alloy and feature integral pre-formed stiffener ribs. The six outer panel segments are made of Carbon-Graphite laminate construction. The Aft Bulkhead contains the switches to operate the Thrust Reverser PDOS. The Aft Bulkhead contains access panels that allow for maintenance.

Engine Attach Flange

The Aft Flange is made of metal alloy forged ring, anodized and primed for corrosion protection. It features 52 holes for attachment bolts to mount the Inlet Cowl to engine fan case and 3 holes for alignment pins.

Inlet Cowl Anti-Icing

The Nacelle anti-ice system provides hot engine bleed air (HPC3) to the titanium lip skin D-section cavity to prevent ice formation. A duct passes between the Inner and Outer barrels to the cyclone ring mounted on the forward bulkhead. The D-section cavity anti-ice air exits overboard through a duct that connects the Aft Bulkhead to the Outer Barrel.



AIR INLET COWL

Power Door Operating System (PDOS)

Location

The Power Door Operating System (PDOS) is located in the nacelle.

Purpose

The purpose of the PDOS system is to power the Fan Cowl Doors and Thrust Reverser / C-ducts opened and closed to gain access to the engine for maintenance.

Description / operation

The PDOS comprises of the following components.

- An electro hydraulic power pack
- Four hydraulic actuators
- Open and close paired control buttons
- One inboard fan cowl position sensor.

The system uses engine oil as an independent hydraulic medium stored in its own internal reservoir

The electrical motor drives the hydraulic power pack to provide the hydraulic muscle power at 2000psi for the opening and closing of the actuators.

The PDOS actuators are located on each engine. There are four actuators, one for each LH / RH Fan Cowl Door and one for each LH / RH Thrust Reverser Half.

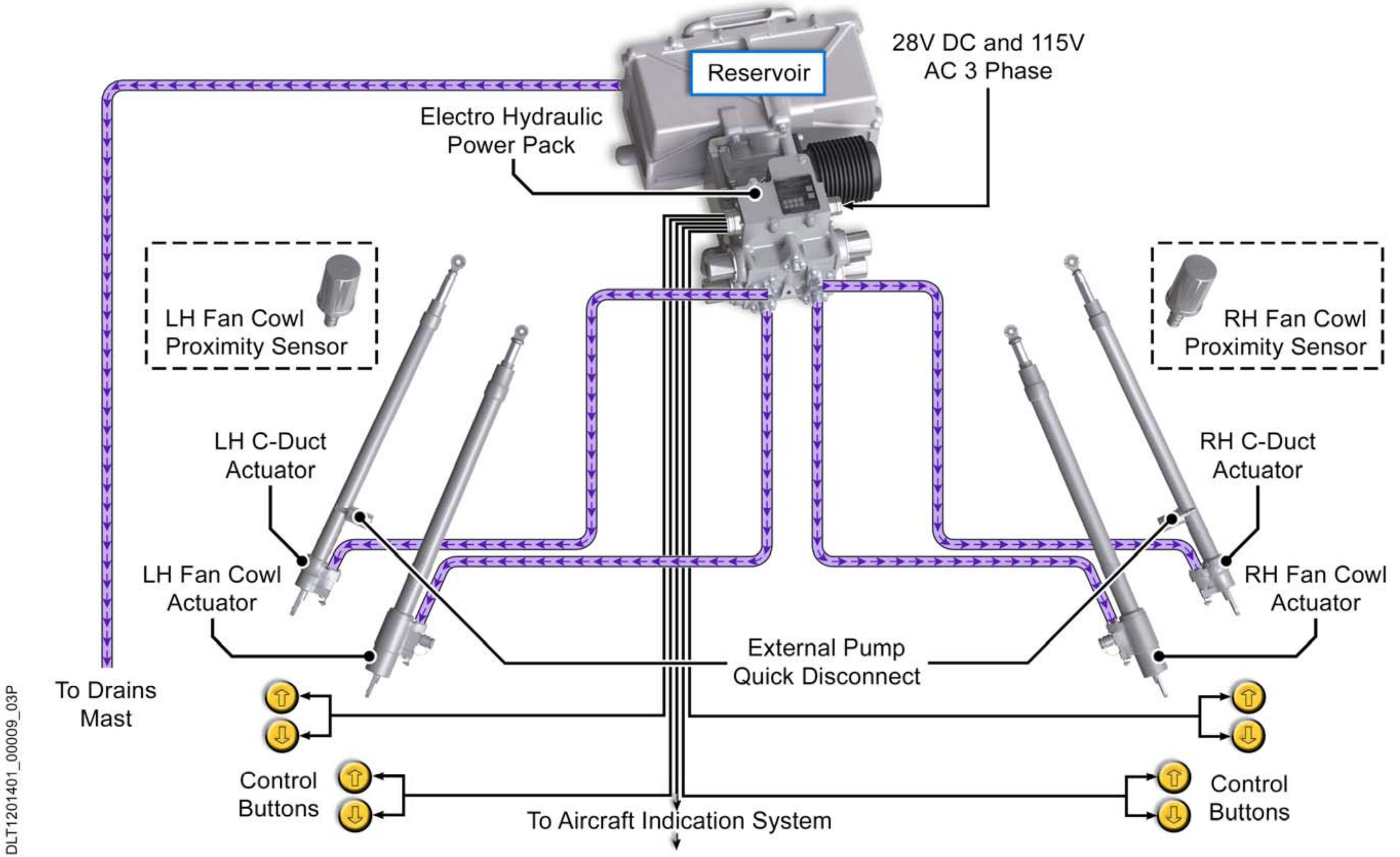
In order to OPEN ⇧ and CLOSE ⇩ the fan cowl doors, the latch access panel is opened and the four latch assemblies have to be released. Once the latches are released the doors are free to be powered. Two yellow arrowed PDOS buttons on each side of the intake cowl marked up and down function the electro hydraulic actuator to OPEN ⇧ or CLOSE ⇩ the individual L/H or R/H fan cowl doors.

By pressing the PDOS ⇧ button, the fan cowl door can be opened to two separate locking positions. The first is 37 degrees and the second 52 degrees, depending on positional access required on the engine. Once in position, the hold open rods display a colour coded band, which confirms the safe condition (green band visible) or unsafe condition (red band visible). The hold open rods lock in position and the PDOS ⇩ button is briefly pressed to release the load off the actuator to the rods.

To close the fan cowl doors, press the PDOS ⇧ button to load the hold-open rods for release. Simultaneously pull and rotate the remote cable handle release and press PDOS ⇩

button to close the fan cowl. The remote cable handle is now released.

The power pack provides a motor over heat signal that is used to prevent the motor from running in over heat conditions.



POWER DOOR OPERATING SYSTEM

Fan Cowl Door Proximity Sensor

Location

The proximity sensor is located on each inboard fan cowl hinge attachment point on the pylon structure.

Purpose

To prevent extension of wing Droop Nose Device (DND) clashing into the inboard fan cowls

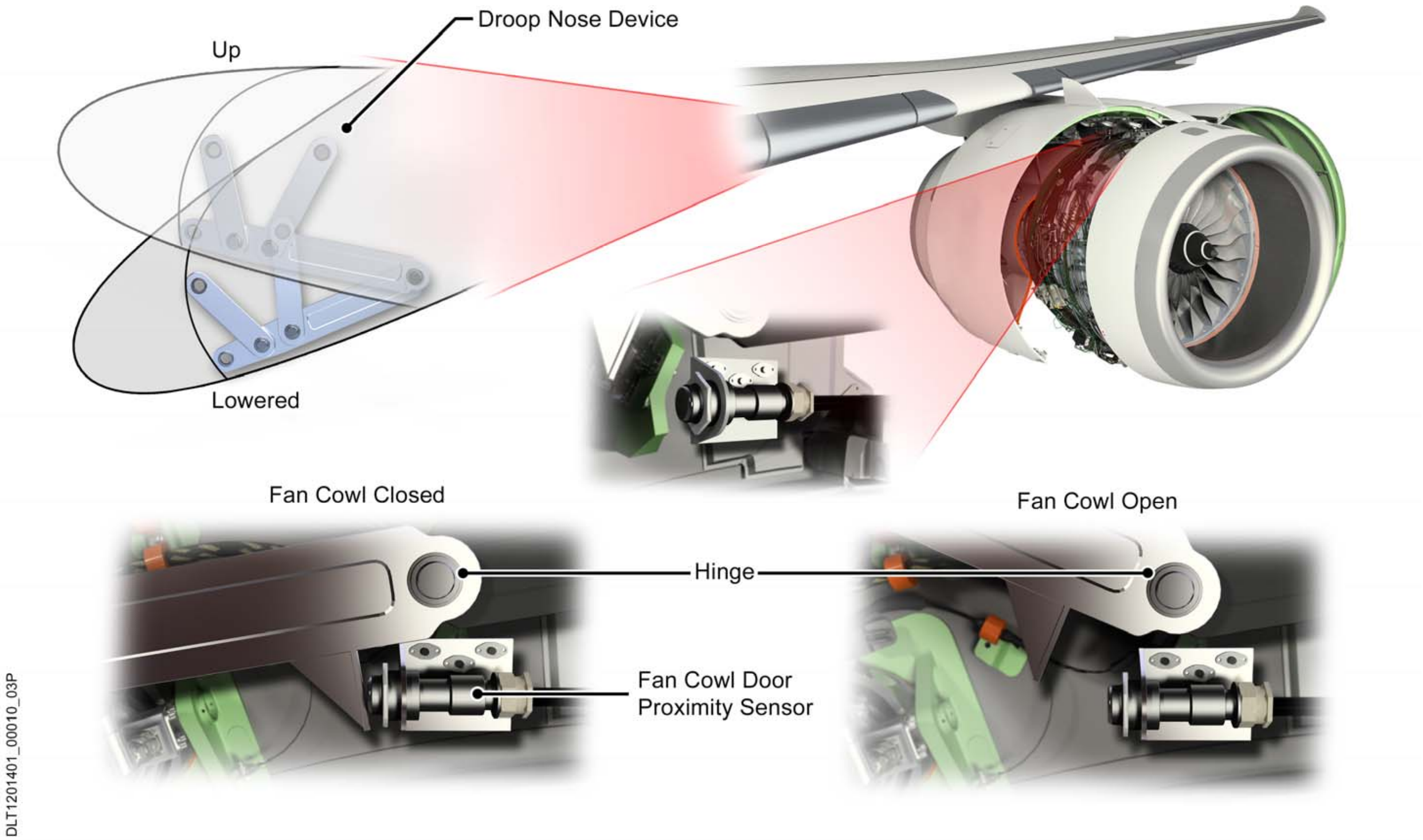
Description

Scenario 1: Inboard DND are retracted

- Fan Cowl door proximity sensor, detects fan cowl completely closed, can be deployed.
- Fan Cowl door proximity sensor detects a non-presence of fan cowl open and sends a message to EIF (Engine Interface Function) DND extension is inhibited with a cockpit indication.

Scenario 2: Inboard DND are extended

- The PDOS is not inhibited.
- Fan Cowl and Thrust Reverser Unit can be opened to intermediate position, but should be aware not to operate the PDOS to fully open position to prevent a clash of cowl with DND.



FAN COWL PROXIMITY SENSOR

Fan Cowl Doors

Location

The two fan cowl doors are located between the air inlet cowl and the thrust reverser.

Purpose

The two fan cowl doors make an aerodynamic protective shell around the engine between the inlet cowl and the thrust reverser. The fan cowl doors give access to the fan case section of the engine and create the outer 'skin' for the zone 1 ventilation.

Description / Operation

The two Fan Cowl Doors (FCD) are manufactured from composite material. Four frame stiffeners on the inner surface of each door, maintain the structural integrity.

Each door has four attachments, the aft three are fixed from the pylon, and the forward hinge is free floating but connected to the opposite fan cowl door via a tie rod.

Four latches secure both doors together at the bottom to close around the LP compressor case and are accessed by opening the Latch Access Panel (LAP). This panel provides aerodynamic smoothness over the latches, as well as acting as a latch baulking device preventing accidental insecurity of the cowl latches.

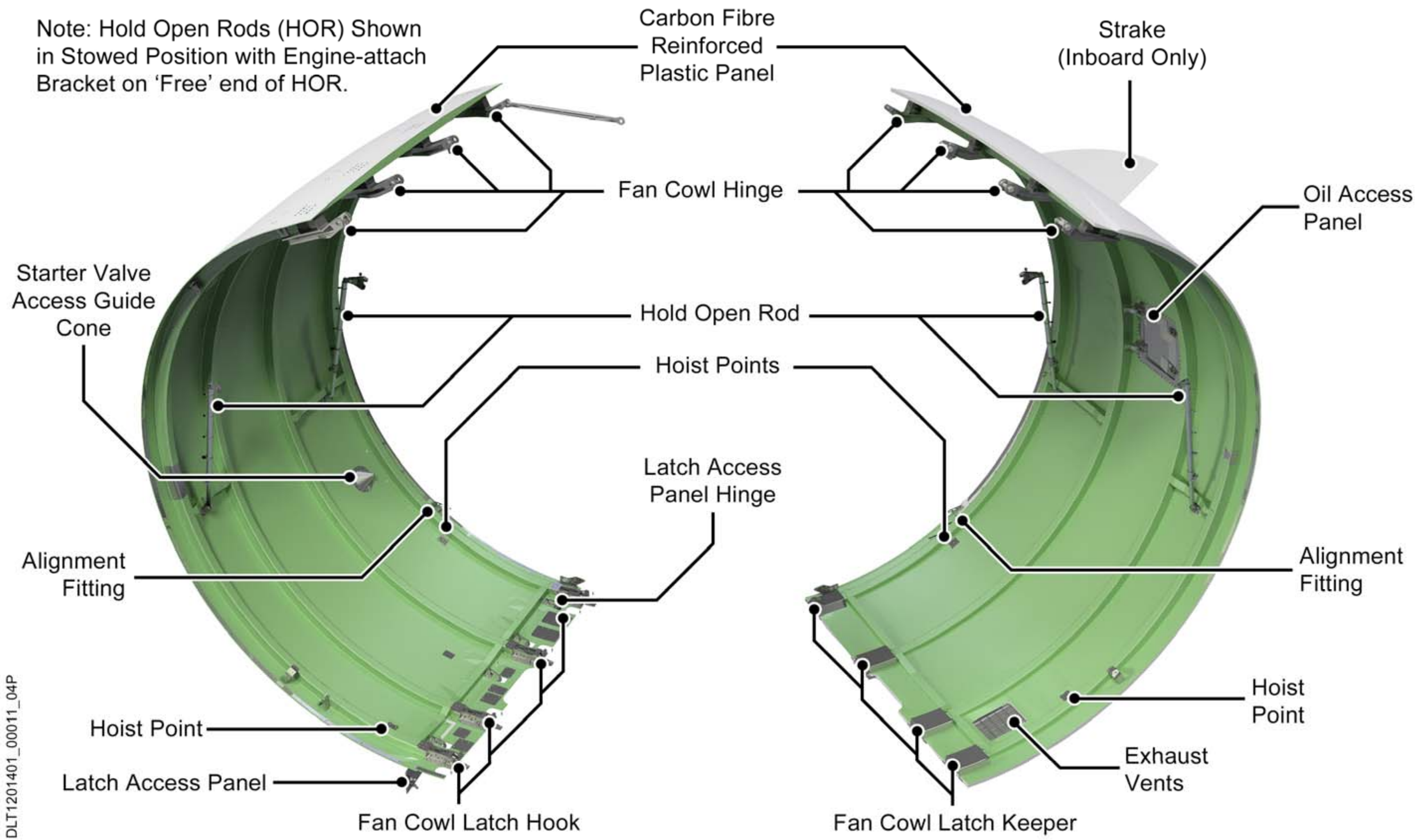
Each door has two permanently attached Hold Open Rods (HOR) these have an automatic locking feature, and remote unlocking device.

Dedicated access panels provide maintenance access for engine oil servicing and starter air valve manual override. An exhaust grill for zone 1 ventilation is located on the R/H fan cowl door.

Each inboard fan cowl has a strake fitted; the strake gives smooth airflow to decrease turbulence. Provisions have been made to install the strake on either door since the doors are interchangeable.

Maintenance Tip

If the PDOS is inoperative the fan cowl doors can be opened manually, but several persons are required to safely manually handle the doors open and close due to the weight and size.



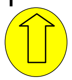
FAN COWL DOORS

Fan Cowl Door Access Operation

Maintenance Practices

The following is a brief description of these tasks. Refer to the Aircraft Maintenance Procedure (AMP) for the full description and the relevant Operational, Maintenance, Storage & Disposal (OMSD) for the correct use of any specialised tooling.

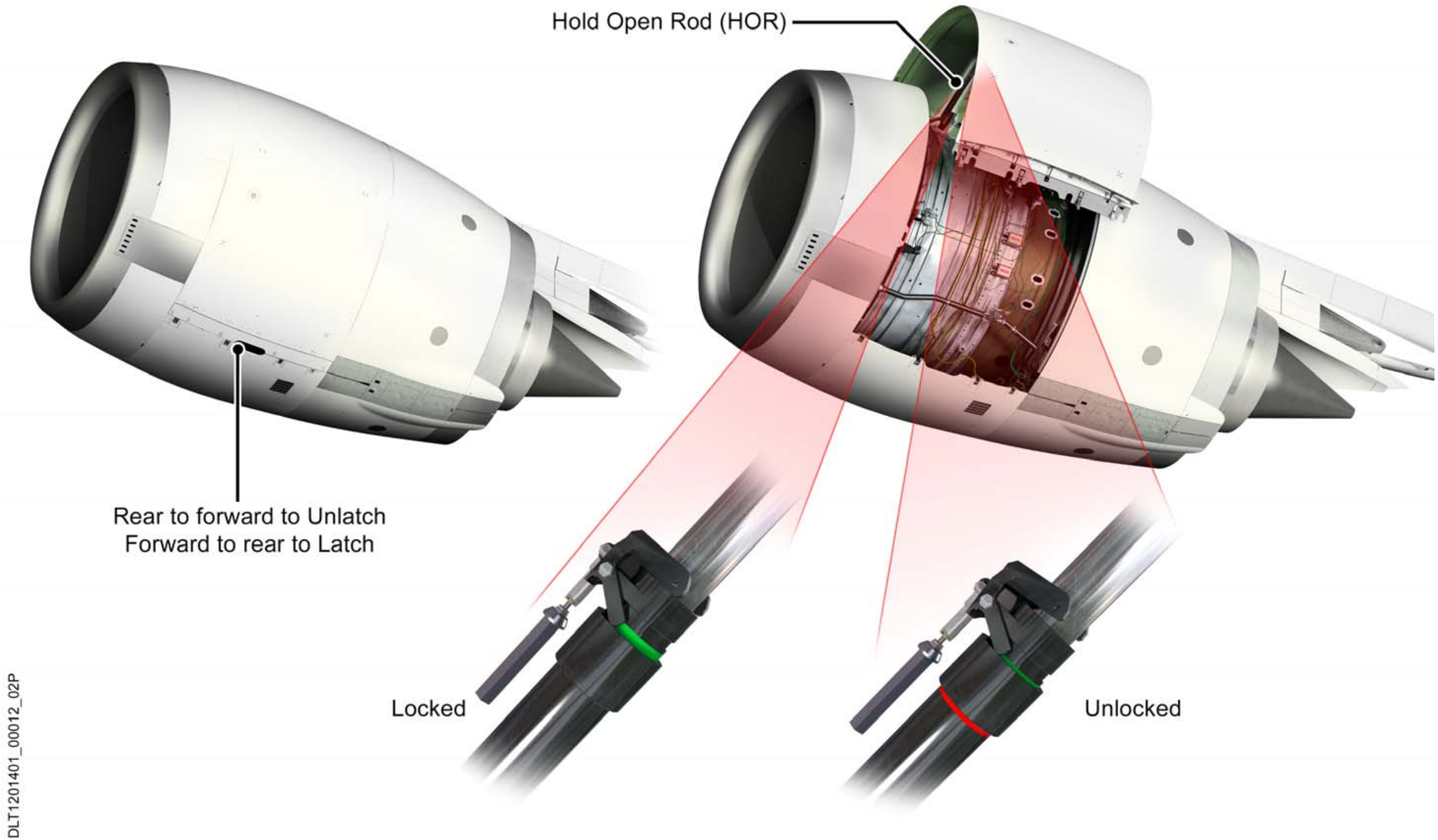
Procedure for opening the doors

- Release the five Latch Access Panel (LAP) catches by pressing the opening buttons.
- Open the four latches that hold the fan cowls closed.
- The four adjustable tension hook latches connect the LH and RH fan cowl doors along the bottom centre line. (The hook part of the latch is on the LH fan cowl door the keeper part of the latch is on the RH fan cowl door).
- Press the  button to activate the PDOS.

CAUTION

IF THE SLATS ARE EXTENDED BEFORE THE FCD IS OPEN, THE PDOS IS NOT INHIBITED. PREVENTION OF FCD CLASHING WITH THE SLATS REMAINS THE JUDGMENT OF THE MECHANICS.

- Monitor the fan cowl opening until the Hold Open Rod safety locks engage. (This is indicated by a green band being visible on the lock assembly). The safety locks automatically engage at the 37º and 52º angles.



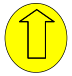
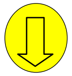
OPENING THE FAN COWL DOORS

Fan Cowl Door Access Operation

Maintenance Practices

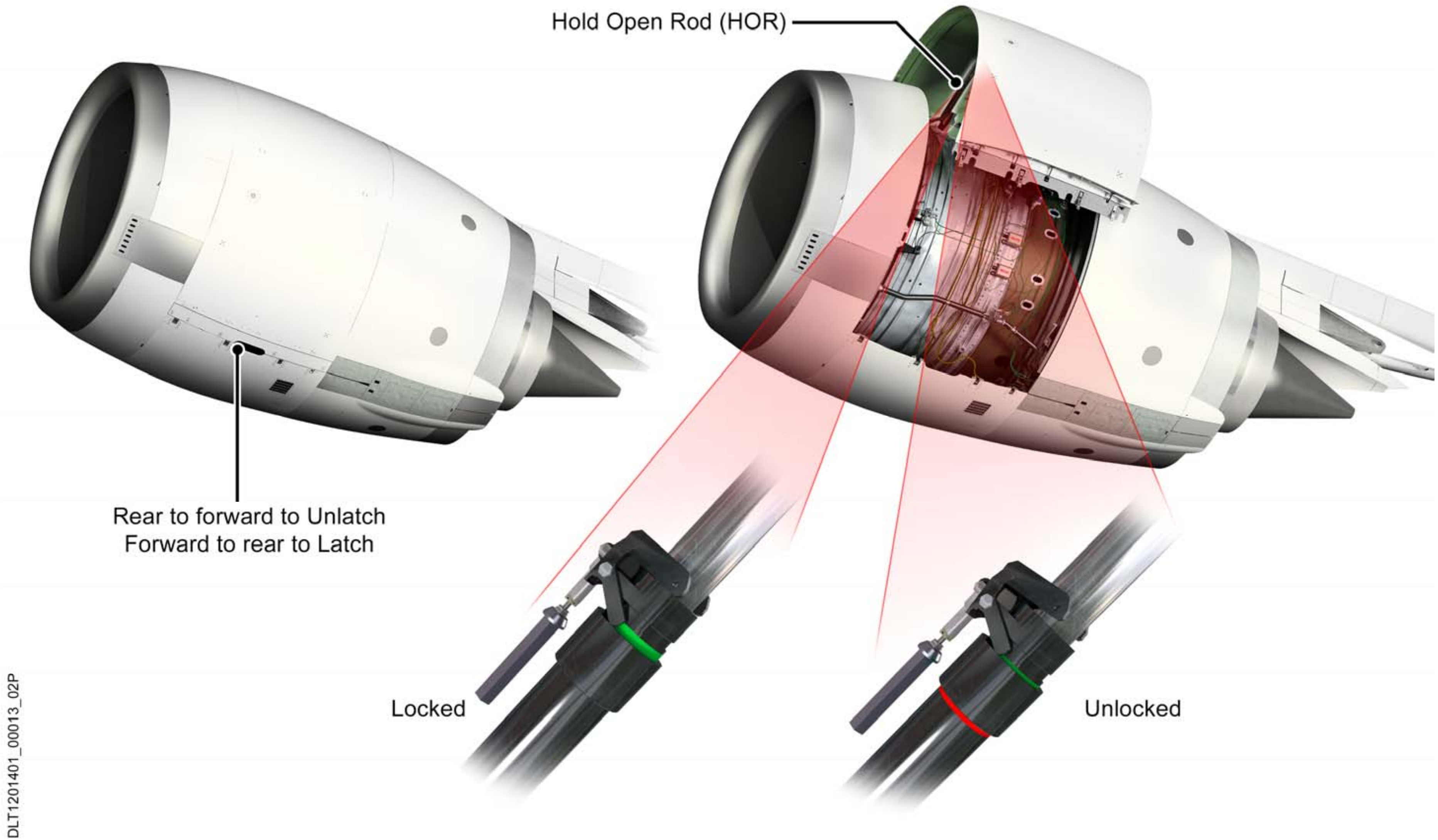
The following is a brief description of these tasks. Refer to the Aircraft Maintenance Procedure (AMP) for the full description and the relevant Operational, Maintenance, Storage & Disposal (OMSD) for the correct use of any specialised tooling.

Procedure for closing the doors

- Activate the remote handle to de-latch the locking collar on the hold open rods.
- Momentary press the  button on the PDOS to release the locking collar on the hold open rod.
- Press the  button to lower the fan cowl.
- When both fan cowls are lowered close the four fan cowl latches.
- Close the Latch Access Panel (LAP) by securing the five catches.
- Ensure that the LAP is secure and flush to the nacelle.

Note

The Latch Access Panel acts as a safety device that prevents the closing of the panel if any fan cowl latches are not correctly installed and locked.



CLOSING THE FANCOWL DOORS

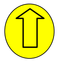
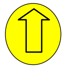
Thrust Reverser / C-duct Access Operation

PDOS is electrically-activated (push-buttons) and hydraulically-actuated (PDOS actuator). Fan Cowl must be opened prior to opening / closing the thrust reverser

Maintenance Practices

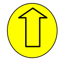
The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Opening the Thrust Reverser/C-Ducts

- Open and secure fan cowls
- Open the thrust reverser Latch Access Panels (LAP)
- Unlatch thrust reverser halves by opening the eight latches. The latch opening and closing sequence must strictly be followed In Accordance With (IAW) the instructions as detailed in the AMP opening and closing procedure.
- Activate Thrust Reverser PDOS  button to open Thrust Reverser
- Un-stow the Thrust Reverser Hold Open Rod (HOR) using a ladder
- Engage the Thrust Reverser Hold Open Rod (HOR) on the engine fitting
- Press the  button and open the reverser to its interim 28.5°or maximum 47° opening angle
- Extend and attach the hold-open rod
- The Hold Open Rod (HOR) has green / red markings for secure / not-secure visual status

C-duct manual opening and closing

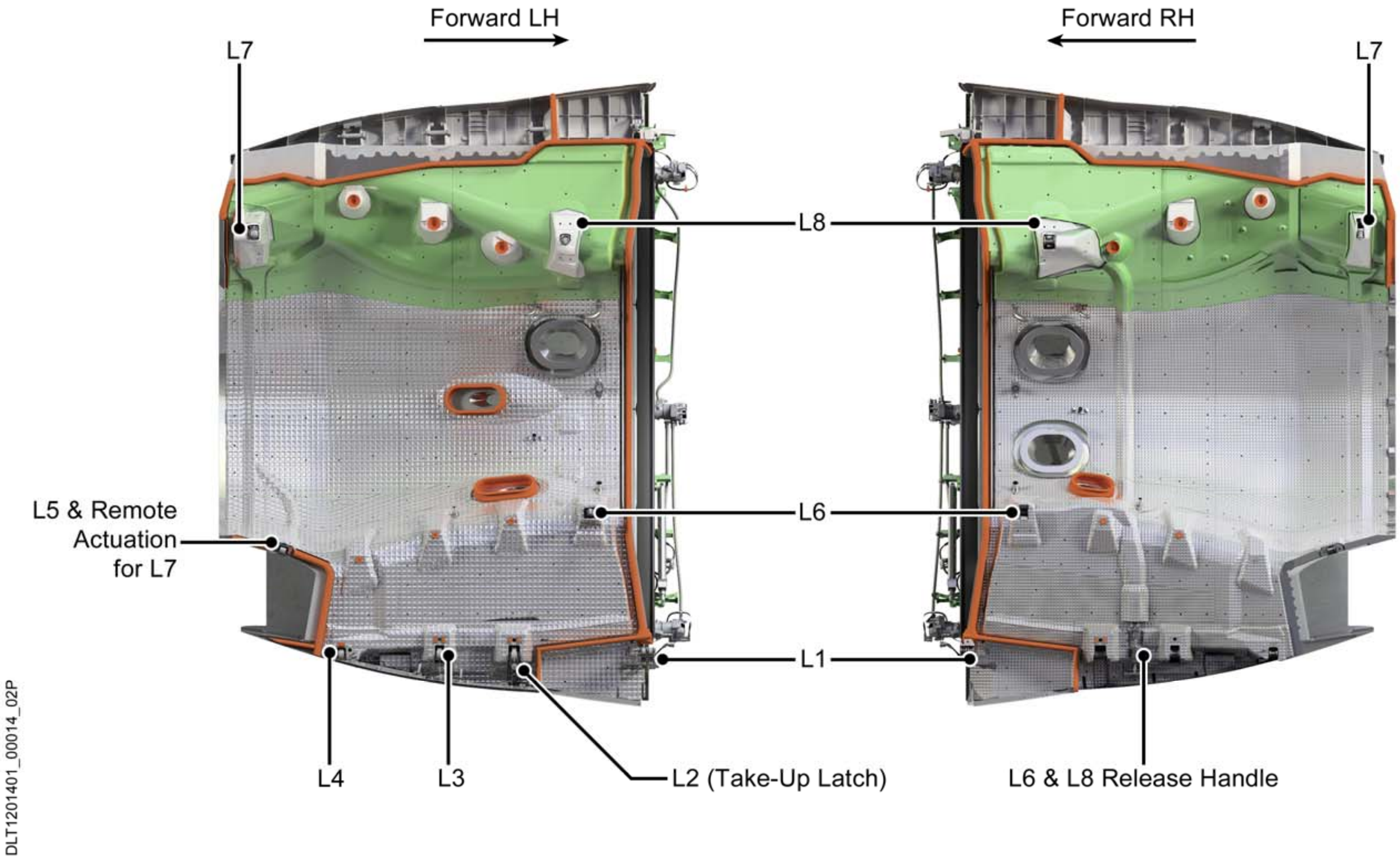
In the event that the PDOS system is inoperative, the thrust reversers can be opened and closed manually with the use of an external hand hydraulic pump (Ground Support Equipment (GSE) containing engine oil), using the quick disconnect valves on the actuator body.

- Open and secure fan cowl, unlatch thrust reverser halves and activate T/R button  to open on the selected T/R half.
- Using a access platform manually un-stow the T/R HOR
- Engage the T/R HOR on the engine fitting. If the T/R HOR is not fully engaged in the engine fitting, the HOR will fall away from the engine.
- Depressurize the PDOS system by momentarily selecting  to load the HOR and secure the thrust reverser half, or proceed to open the reverser to its maximum opening angle.

OPENING THE THRUST REVERSER / C-DUCTS

Thrust Reverser Latches

- Four lower track beam latches (Latches 1, 2, 3, 4)
- Two upper bifurcation latches (Latches 7, 8)
- One lower bifurcation latch (Latch 6)
- One Aft Core Cowl latch (Latch 5)



THRUST REVERSER LATCH POSITIONS

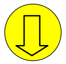
Thrust Reverser / C-Duct Access Operation

PDOS is electrically-activated (push-buttons) and hydraulically-actuated (PDOS actuator). Fan Cowl must be opened prior to opening / closing the thrust reverser.

Maintenance Practices

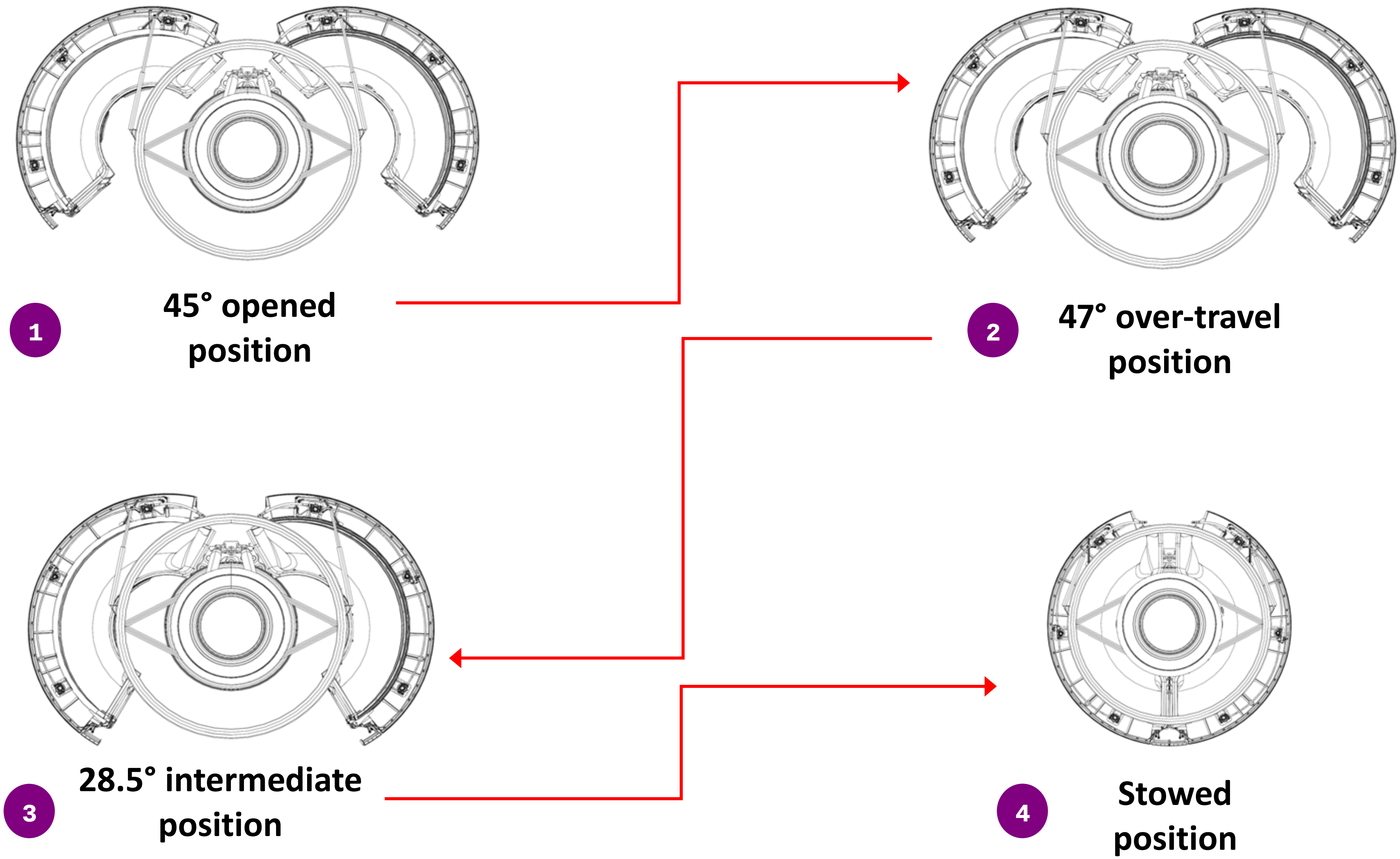
The following is a brief description of these tasks. Refer to the Aircraft Maintenance Procedure (AMP) for the full description and the relevant Operational, Maintenance, Storage & Disposal (OMSD) for the correct use of any specialised tooling.

Closing the Thrust Reverser/C-Ducts

- Press the PDOS  button (this relives the load on the safety locking latch).
- Disengage the Thrust Reverser Hold Open Rod (HOR) on the engine fitting.
- Prior to closing Thrust Reverser using PDOS ensure the HOR is collapsed and stowed within the attach clip.
- Press the PDOS  button.
- Initially using latch two draw the C-duct half's together close the ducts and secure the remaining seven latches in the sequence illustrated in the AMP procedure.
- Close the thrust reverser Latch Access Panel (LAP)
- Close and secure fan cowls

The Figure below shows the T/R in four stages of opening / closing:

1. Fully open position (45°)
2. Over-rotated position (47°)
3. Intermediate position (28.5°)
4. Fully closed



THRUST REVERSER/C-DUCT CLOSING OPERATION

Thrust Reverser

Location

The thrust reverser assembly is located at the rear of the fan cowl doors and encompasses the engine core.

Purpose

The purpose of the thrust reverser is to assist the aircraft braking system on landing by directing the LP compressor bypass airflow from a rearward to forwards reaction thrust.

Description

The thrust reverser structure is made of two halves called C-ducts. Each C-duct is attached to the pylon by four hinges two fixed and two floating and is held together by a total of eight latches. Four lower track beam latches, two upper bifurcation latches, one lower bifurcation latch and one core cowl latch.

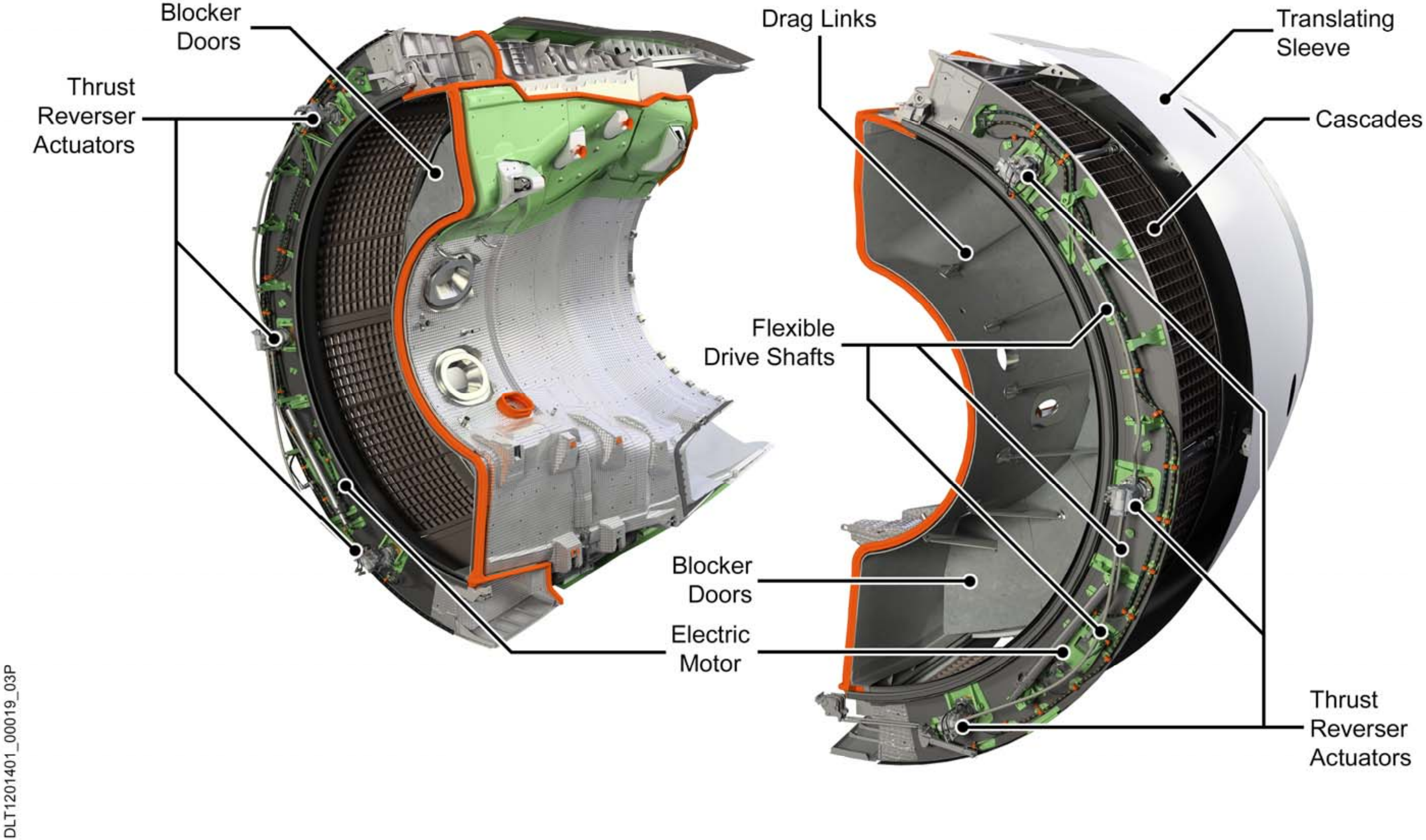
Each C-duct has a translating sleeve, which deploys rearwards when commanded by the aircraft system. A single electrical motor operates three ball screw actuators for each translating cowl through Flex Synchro Shafts which are all connected to the centre actuator.

The upper and lower actuators have locking mechanisms and the centre actuator is a non-locking actuator. The upper actuator on the L/H side and the lower actuator on the R/H side provide the positional feedback to the Electronic Thrust Reverser Actuation Controller (ETRAC). The thrust reverser operation is initiated by manual selection of the cockpit reverse thrust levers.

As the translating sleeves deploy rearwards, drag links pull six composite blocker doors per half into the bypass duct gas path. The blocker doors change the direction of the LP bypass airflow from rearwards to a forward direction through eight composite cascade panels.

Though manually selected, positional control of each thrust reverser assembly is by the ETRAC (one per engine), interfacing with the aircraft Engine Interface Function (EIF) Core Processing Input / Output Module (CPIOM).

The Engine Electronic Controller EEC monitors the position of the translating sleeves for flight deck indication, translating sleeve lock position and control of the thrust as demanded by the reverse thrust lever positioning.



THRUST REVERSER ‘C’ DUCTS

Thrust Reverser Operation

Location

The Electric Thrust Actuation System Controller (ETRAC) computers are installed in the forward cargo compartment.

The electrical motors and ball screw actuators of the thrust-reverser system are on the thrust-reverser / C-duct structure.

Description / Operation

To assist the wheel brake system in slowing the aircraft during landing or rejected take-off, the thrust-reverser system with two translating sleeves is used.

The Engine Interface Function (EIF) receives the position of the thrust reverser lever and sends deploy / stow command signals to the ETRAC.

The ETRAC controls the thrust-reverser movement and locking

The EEC controls the engine speed (N1) dependent on the Thrust Reverse lever position. (Min = Idle, Max=70%N1)

Control and Indicating

The operation of the thrust-reverser (T/REV) is shown on the Control & Display System (CDS) Primary Engine Display. The primary thrust indicator shows the thrust reverser position and thrust setting.

Deployment Sequence

1. Throttle lever in idle position and radio altimeter within 6ft of ground (80-320 ms delay for PRIM to command Tertiary (Track) Lock release).

2. PRIM commands track lock release. (80 ms delay for Tertiary (Track) Lock to release). (200 ms delay for EIF1 & 2 to receive confirmation of Tertiary (Track) Lock release).

3. Throttle lever in rev idle position and Weight on Wheels.

(3 X 200 ms delays for EIF2 to send deploy command).

4. EIF2 deploys command received by ETRAC.

(500 ms delay for EIF1 and power system to enable 230 Vac).

5. 230 Vac Supplied to ETRAC. (ETRAC requires Deploy AND 230 Vac to command lock release). (500 ms delay for ETRAC DC link to reach useable voltage).

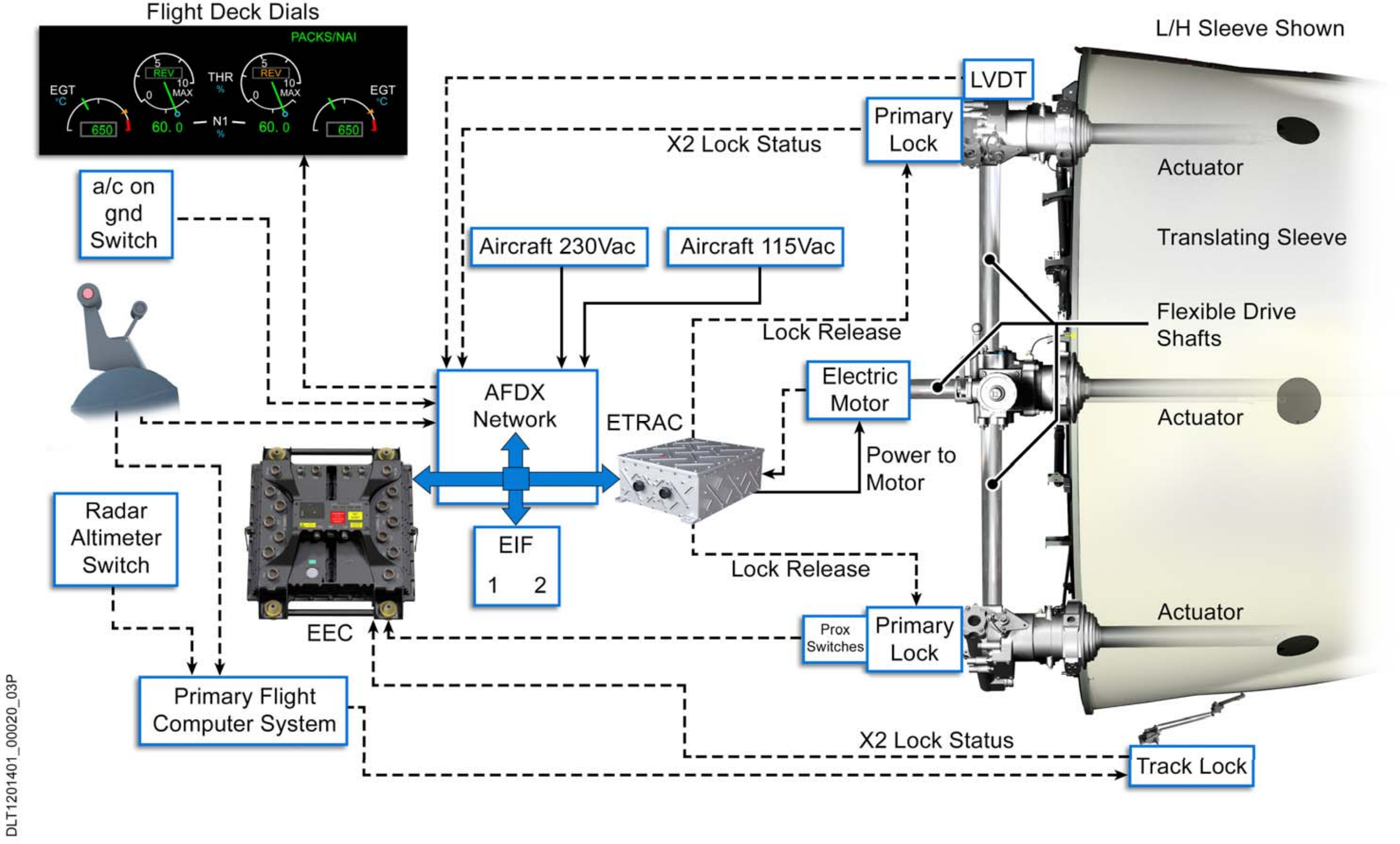
6. ETRAC commands release of primary locks (100 ms delay for primary locks to release).

7. ETRAC powers motor (100 ms delay for ETRAC to detect Fully Deployed Position and power down motors and solenoids).

8. RVDT indication equivalent to > 85 % of TRAS position reached.

9. EEC modulates rev engine thrust against thrust reverse lever position.

10. Fully Deployed Position achieved.



THRUST REVERSER DEPLOY

Stow Sequence

1. Thrust reverse lever NOT selected to rev (230 VAC applied to ETRAC) (PFCS commands Tertiary (Track) Lock release) (300 ms delay for EIF2 to send stow command).

2. EIF2 stow command received by ETRAC.

(Start of TRAS stow) (ETRAC powers motor).

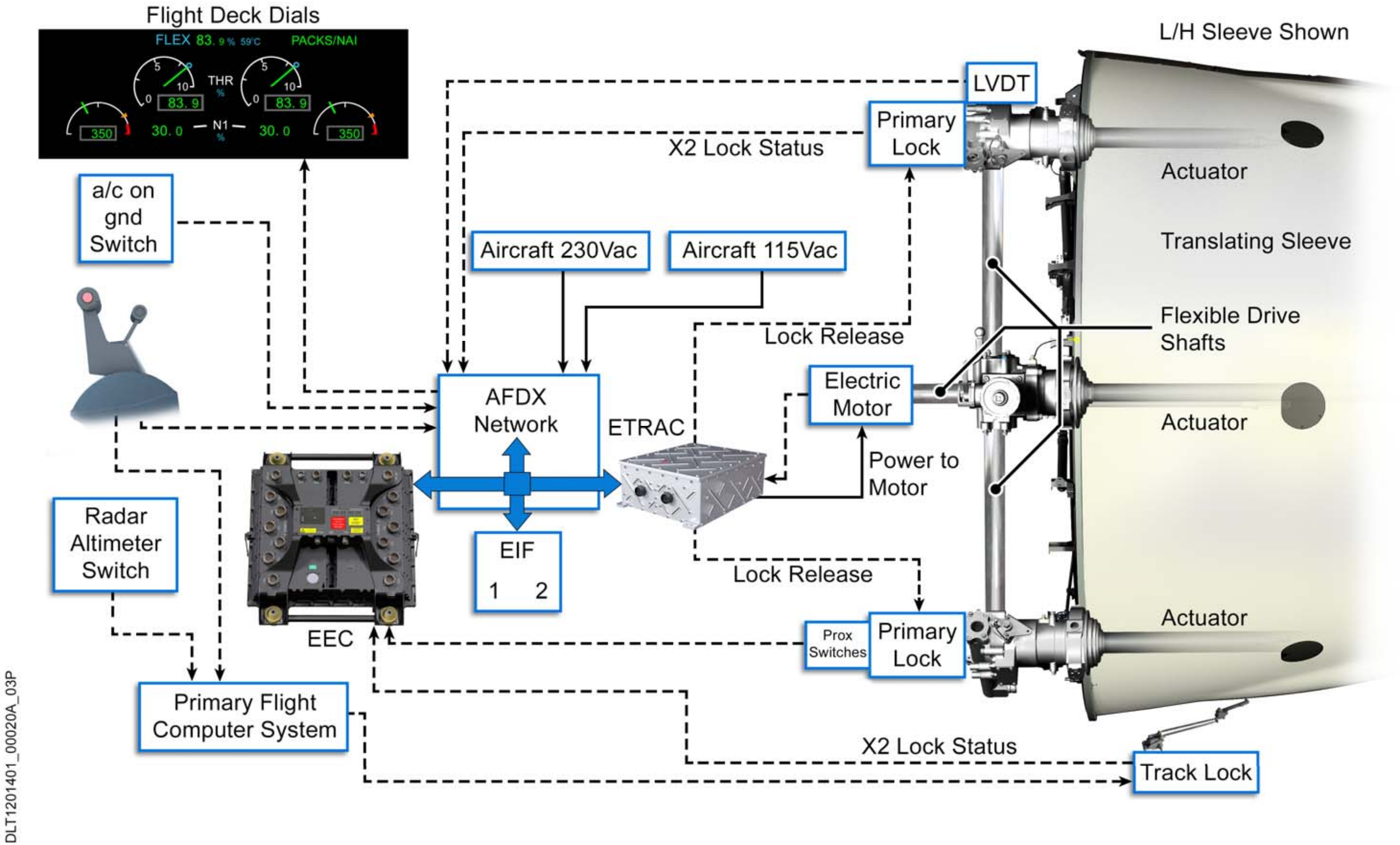
3. Tertiary (Track) Lock engaged.

4. Primary Locks engaged.

(100 ms delay for ETRAC to detect stowed condition).

5. ETRAC detects stowed condition.

(3 second delay for EIF1 and power system to remove 230 Vac).



THRUST REVERSER STOW

Thrust Reverser (T/R) lockout for dispatch in-op T/R and Lock out for Maintenance.

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMM for the full description and the relevant OMSD for the correct use of any specialised tooling.

Deactivation for Maintenance

Deactivation of the T/R for maintenance is carried out as listed below:

- Electrical deactivation:

- Electrical deactivation through the on board Maintenance System (OMS) Power Distribution Monitoring and Maintenance Function (PDMMF) commands the Electronic Power Distribution System (EPDS) to inhibit the electrical power sent to the ETRAC. This step is sufficient, if maintenance is done on the engine and not on the T/R system.

Deactivation for Dispatch

Deactivation of the T/R for flight dispatch is carried out in two main steps:

- Electrical deactivation:

- An electrical deactivation through the OMS (PDMMF) commands the EPDS to inhibit the electrical power sent to the ETRAC.

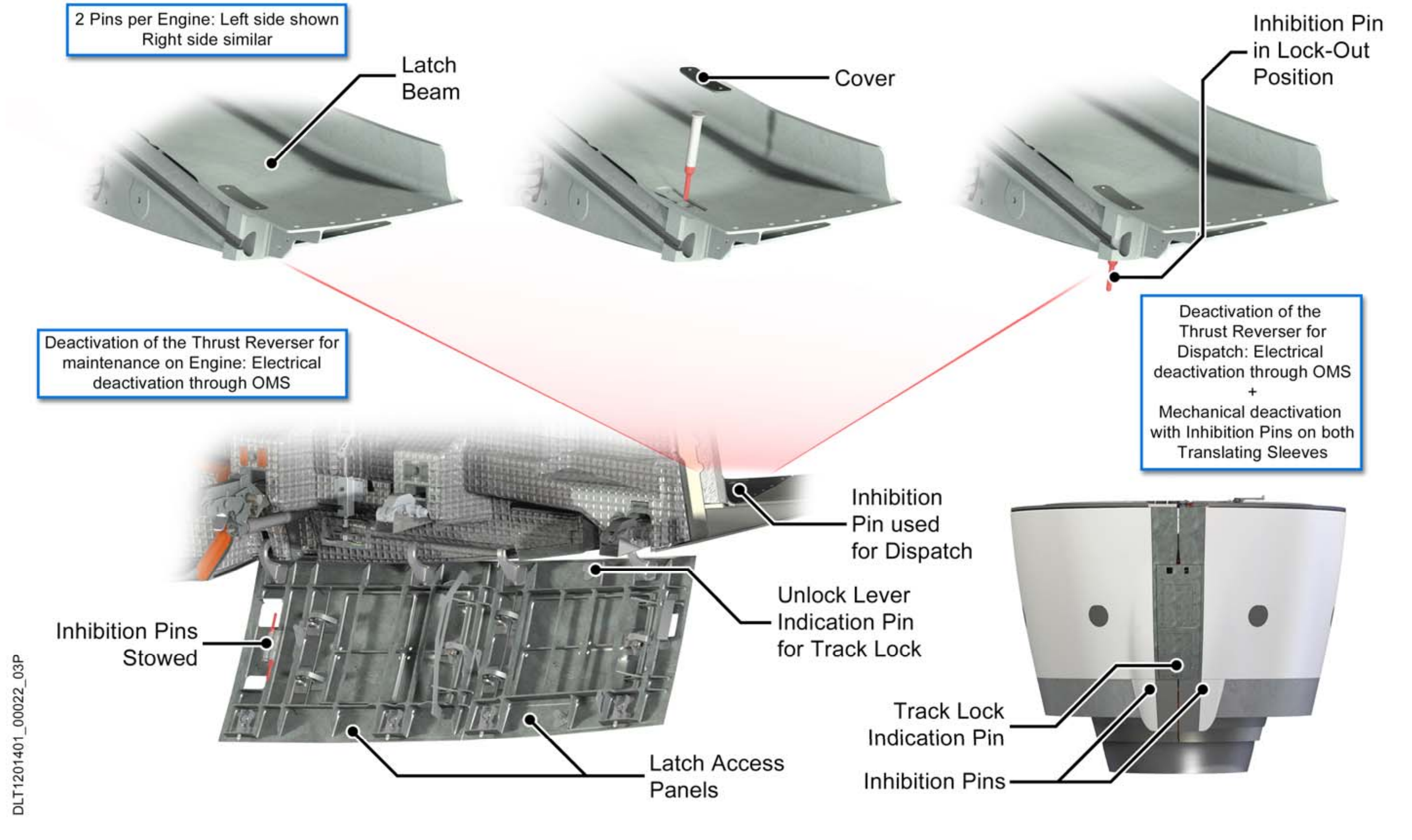
- Mechanical deactivation:

- Deactivation pins are located and stored inside the latch access panel (when not in use for deactivation). These inhibition pins are installed on the aft end of the latch beams to mechanically inhibit the T/Rev sleeve movement. In this position each pin locks the lower translating sleeve slider in the stowed position and is visible externally. (protruding red flexible indicator).

Maintenance Tip

The unlock lever indication pin shows that the track lock is in maintenance position (unlocked) is installed on the T/R latch beam aft access panel.

The inhibition pins installed during the deactivation for dispatch are located just outside the aft access panel. Make sure that the correct pin is identified.



THRUST REVERSER LOCKOUT FOR DISPATCH INOP T/R

Manual operations of T/R stow / deploy

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMM for the full description and the relevant OMSD for the correct use of any specialised tooling.

There is a track lock installed on each latch beam this includes a hook and a manual unlock lever.

A protective cover must be open to access the unlock lever.

The lever is manually moved from the "lock active" position to the “maintenance" position.

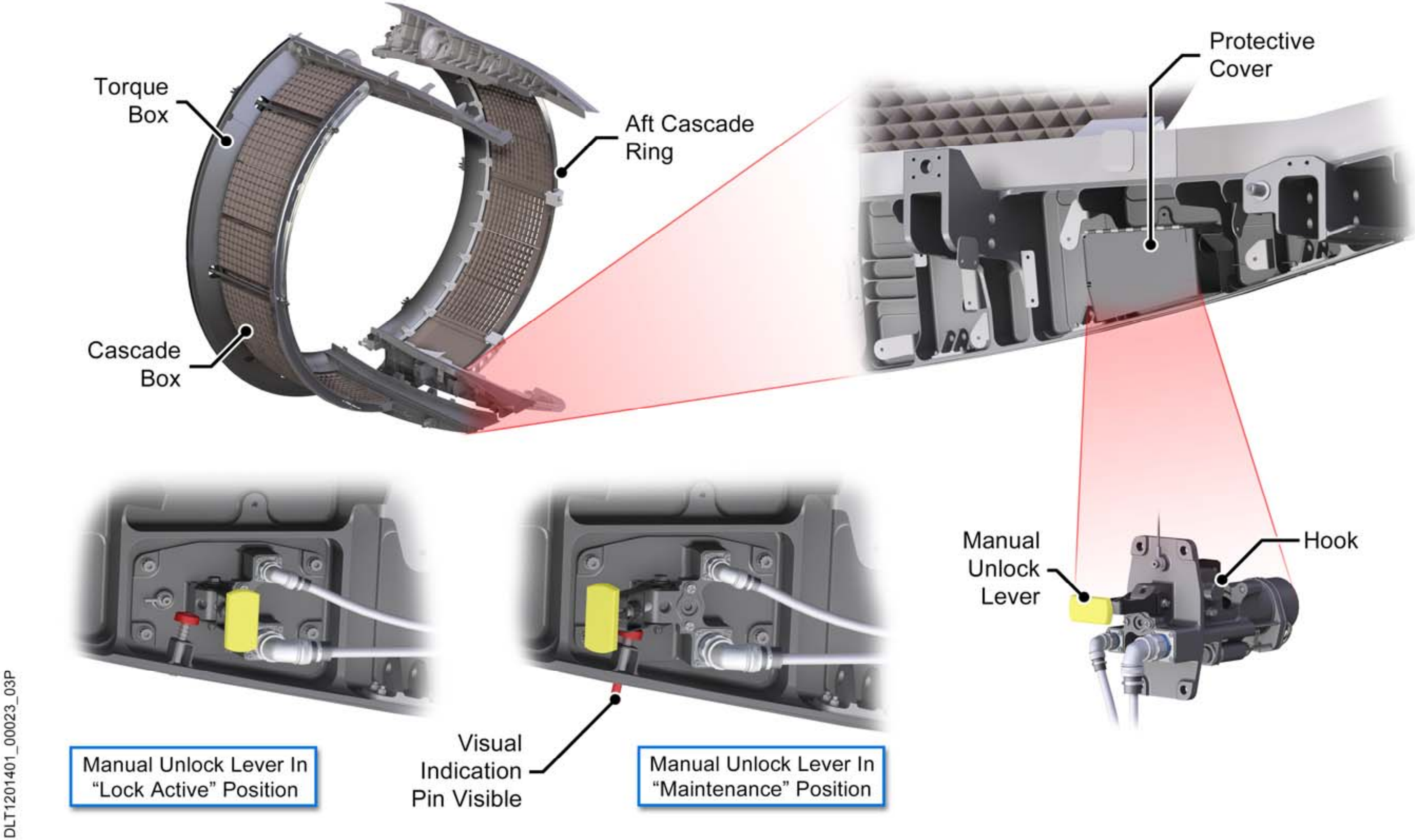
In the "maintenance" position, the lever pushes on a visual indication pin, when the latch access panel is closed. The indication pin protrudes outside the latch access panel, giving indication that the track lock has been left in the "maintenance" position.

A standard square drive tool is required on the centre actuator to stow or deploy the T/Rev translating sleeves.

Cascade Panels

The cascade panels installed between the torque box and the aft cascade ring divert the fan airflow in forward or side direction when the translating sleeve is deployed.

There are different types of cascade panels which may or may not be Interchangeable IAW the AMP.



MANUAL OPERATIONS OF T/R STOW / DEPLOY

Hot Gas Exhaust System

Location

The hot gas exhaust system is located at the rear of the engine.

Purpose

The purpose of the hot gas exhaust system is to:

- Reduce the noise levels of the hot gases exiting the core of the engine.
- Provides correct nozzle gas area to maintain engine performance.
- Provides an interface between the hot and cold exiting gases.

Description

The hot gas exhaust system comprises of three assemblies:

- The outer exhaust nozzle.
- The forward centre body exhaust nozzle.
- The aft centre body exhaust nozzle.

The nozzle formed by these three assemblies provide the aerodynamic control of engine core flow, while optimising thrust, acoustic performance and tail bearing housing thermal protection.

Exhaust Nozzle

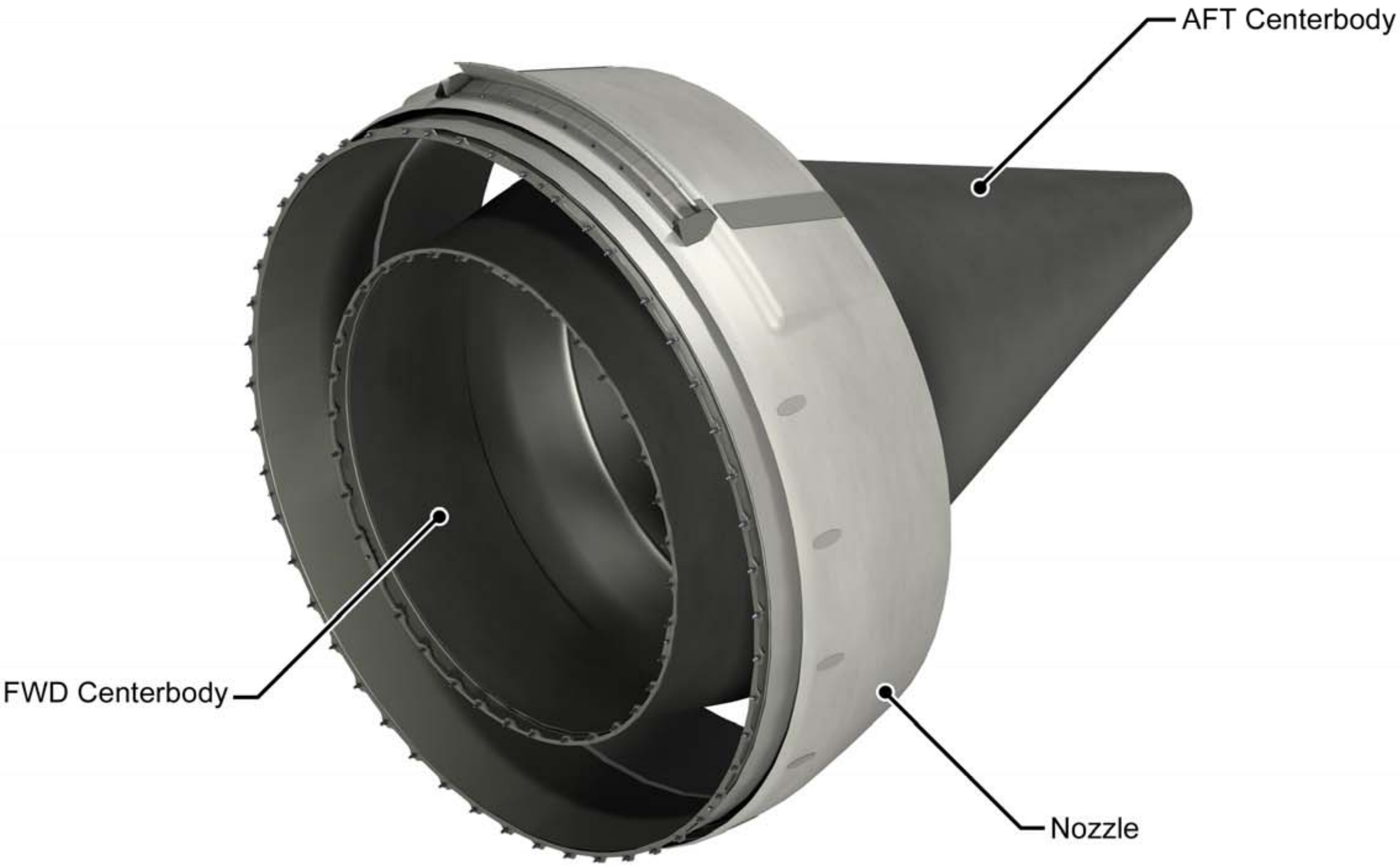
The exhaust nozzle is attached to the rear flange of the Tail Bearing Housing (TBH) support structure, which is part of the LP turbine module and forms the outer annulus of the hot gas exhaust system. A metal finger fire seal provides fire zone integrity for zone 3, isolates the forward pylon cavity and shields the aft pylon from direct exhaust gas impingement. Cross flow seals provide aerodynamic sealing and noise reduction to the aft pylon interface.

Forward Exhaust Centre body

The forward exhaust centre body is attached to the rear flange of the TBH and together with the aft centre body forms the inner annulus of the hot gas exhaust system.

Aft Exhaust Centre body

The aft exhaust centre body is attached to the rear face of the forward centre body and together forms the inner annulus of the hot gas exhaust system.

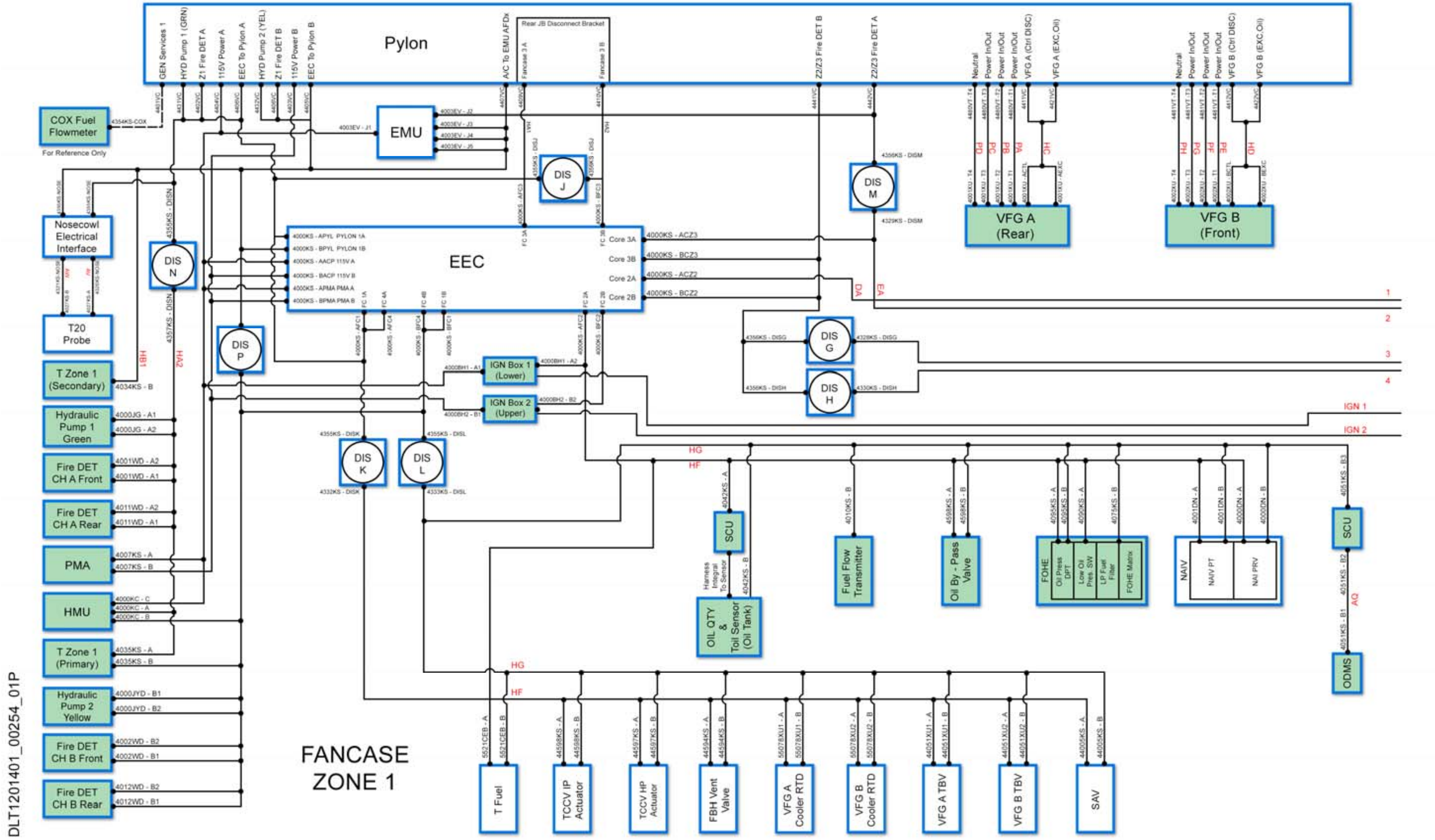


HOT GAS EXHAUST SYSTEM

Nacelle Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.

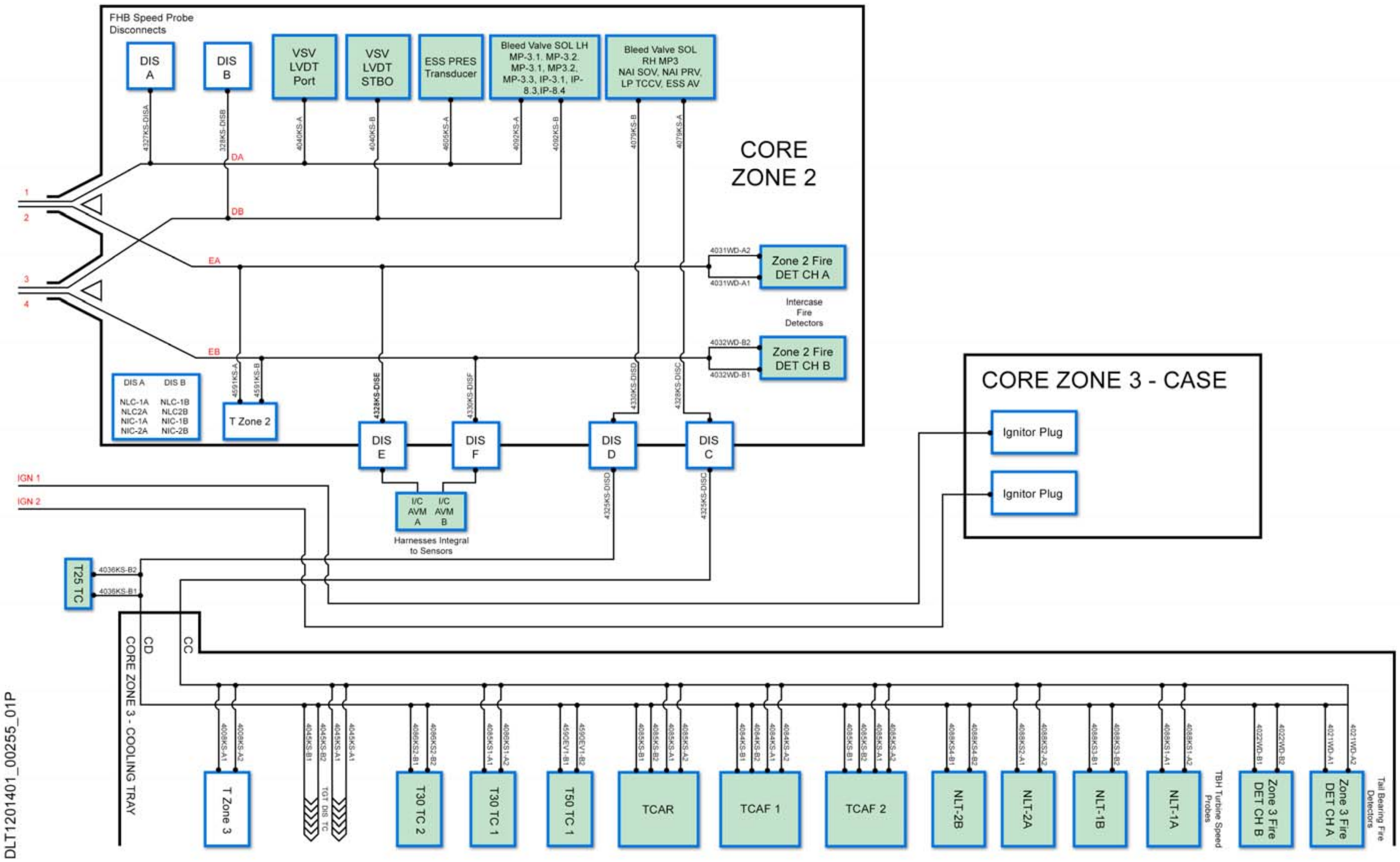


ZONE 1 WIRING DIAGRAM

Nacelle Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.



ZONE 2 & 3 WIRING DIAGRAM

Section 2 - Nacelle

Objectives

At the end of this section the student will be able to:

- State the purpose of the nacelle as fitted to the Trent XWB engine/A350.
- Locate and identify the major assemblies that form the nacelle of the Trent XWB engine/A350.
- Describe the purpose and operation of the major assemblies that form the Trent XWB engine/A350.
- State the WARNINGS AND CAUTIONS associated with the nacelle as fitted to the Trent XWB engine/A350.
- Describe how the Trent XWB nacelle interfaces with other engine and aircraft systems.

End of Nacelle Section

Section 3 – Propulsion System

Section 3 – Propulsion System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance procedural level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the propulsion system.
- Locate and identify the major assemblies that form the propulsion system.
- Describe the purpose and operation of the major assemblies that form the propulsion system.
- State the WARNINGS & CAUTIONS associated with the propulsion system.
- Describe how the propulsion system interfaces with the aircraft and other airframe systems.

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Engine Attachments

Location

The engine mounts attach to the engine at two locations.

The front mount at the top of the fan case front mount ring. The rear mount located on the top of the tail bearing housing support structure.

Purpose

The purposes of the engine mounts are to support the weight of the engine, transmit thrust loads to the aircraft pylon, prevent rotation of the engine about its axis and support lateral loading.

The mounts are designed to maintain attachment integrity if an attachment bolt or link fails the attachment is not compromised.

Description

The two assemblies that attach the engine to the pylon are:

Front Mount

The front mount is made up of five main components; they are three link arms, the main body and several spherical bearings. It is located at the top of the LP compressor case and attached to the aircraft pylon through the main body of the mount by three tension bolts. The bolts are locked together with lock plates and a support bracket. The main body has provision for the attachment of three suspension links; these links go from the main body of the mount to the engine; these are attached to the engine and main body by pins with safety bolts and washers to secure them. The bolts go through the centre of the pin and washer to lock it in position. The suspension links are floating and allow for expansion and contraction due to thermal and stress loads from the engine.

The front mount transmits the following loads:

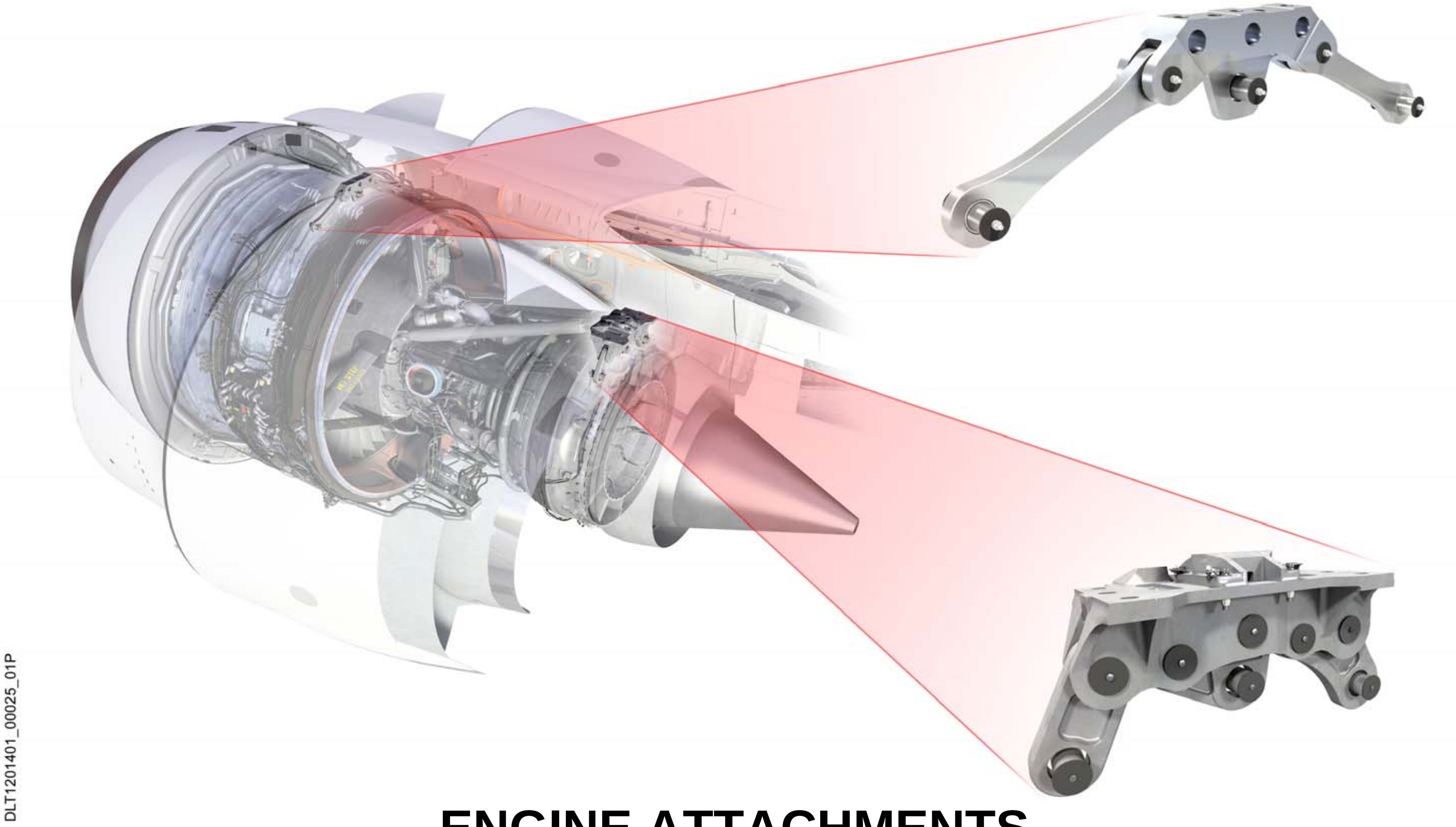
- Vertical.
- Side.

Rear Mount

The rear mount is made up of seven main components these are three suspension links, a main body, a clevis bracket, several spherical bearings and a floating lug casting that attach the thrust struts to the main body. The mount is located at the tail bearing housing support structure, which is part of the LP turbine module. The rear mount attaches to the aircraft pylon by six tension bolts and barrel nuts, several locking plates and brackets are used to safety the bolts. The engine is attached to the main body of the mount by floating suspension links with spherical bushes at each end these are secured into position on both ends by pins with safety bolts and washers. These bolts go through the centre of the pin and washers to lock it in position. The suspension links are floating and allow for expansion and contraction due to thermal and stress loads from the engine.

The rear mount transmits the following loads:

- Vertical.
- Side.
- Torsion.
- Thrust.



ENGINE ATTACHMENTS

Thrust Struts

Location

Two thrust struts are attached between lugs on the engine intermediate case and the rear mount.

Purpose

The purpose of the thrust struts is to transmit the thrust developed by the engine to the rear mount, then subsequently to the airframe.

Description

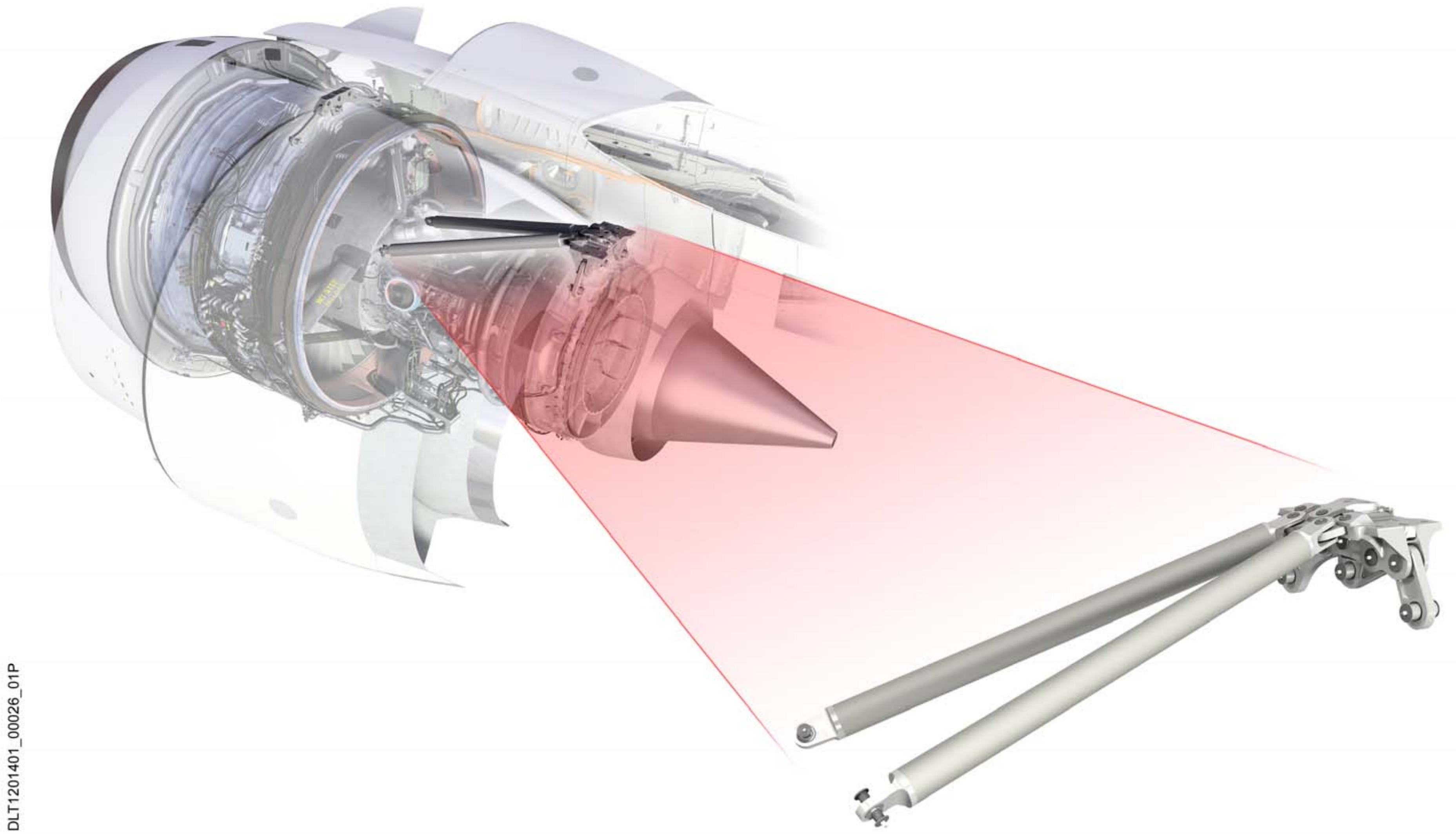
The thrust strut tubes are manufactured from titanium and have an attachment lug at the front end and a clevis end attachment at the rear; these are welded to the end of each thrust strut.

When the intermediate case is manufactured, two clevis end attachments are machined from the billet of material that the case is made from.

The thrust strut front lug is attached to the intermediate case clevis ends with attachment pins that have safety bolts and washers to secure them.

The thrust strut rear clevis end is attached to the rear mount lug via a pin and washer with a safety bolt.

The rear mount lug is allowed to ‘free float’ on the rear mount clevis. A single piece casting is attached to the mount clevis by 3 pins with safety bolts and washers to secure them.



ENGINE THRUST STRUTS

Engine Bleed-Air System (EBAS)

Introduction

The function of the Engine Bleed Air System (EBAS) is to supply the users with pressure and temperature controlled air from the engines.

Location

The EBAS components associated with the engine are located on the engine core and within the pylon.

Description

The EBAS utilises two separate engine air off-takes, IPC stage 8 or HPC stage 6. Only one off-take is used at one time, off-take usage selection is dependent on air system demand, engine RPM & duct pressure.

Two airframe mounted Core Processing Input / Output Modules (CPIOMs) host a Bleed Air System (BAS) application. The BAS application manages the EBAS through the Avionics Data Communication Network (ADCN).

EBAS LRUs

The engine core Zone LRUs are as follows:

- High Pressure Valve (HPV)
- Intermediate Pressure Check Valve (IPCV)
- Manifold Pressure Valve (MPV)
- Fan Air Valve (FAV)

Pylon Zone LRUs:

- Over Pressure Shut-Off Valve (OPSOV)

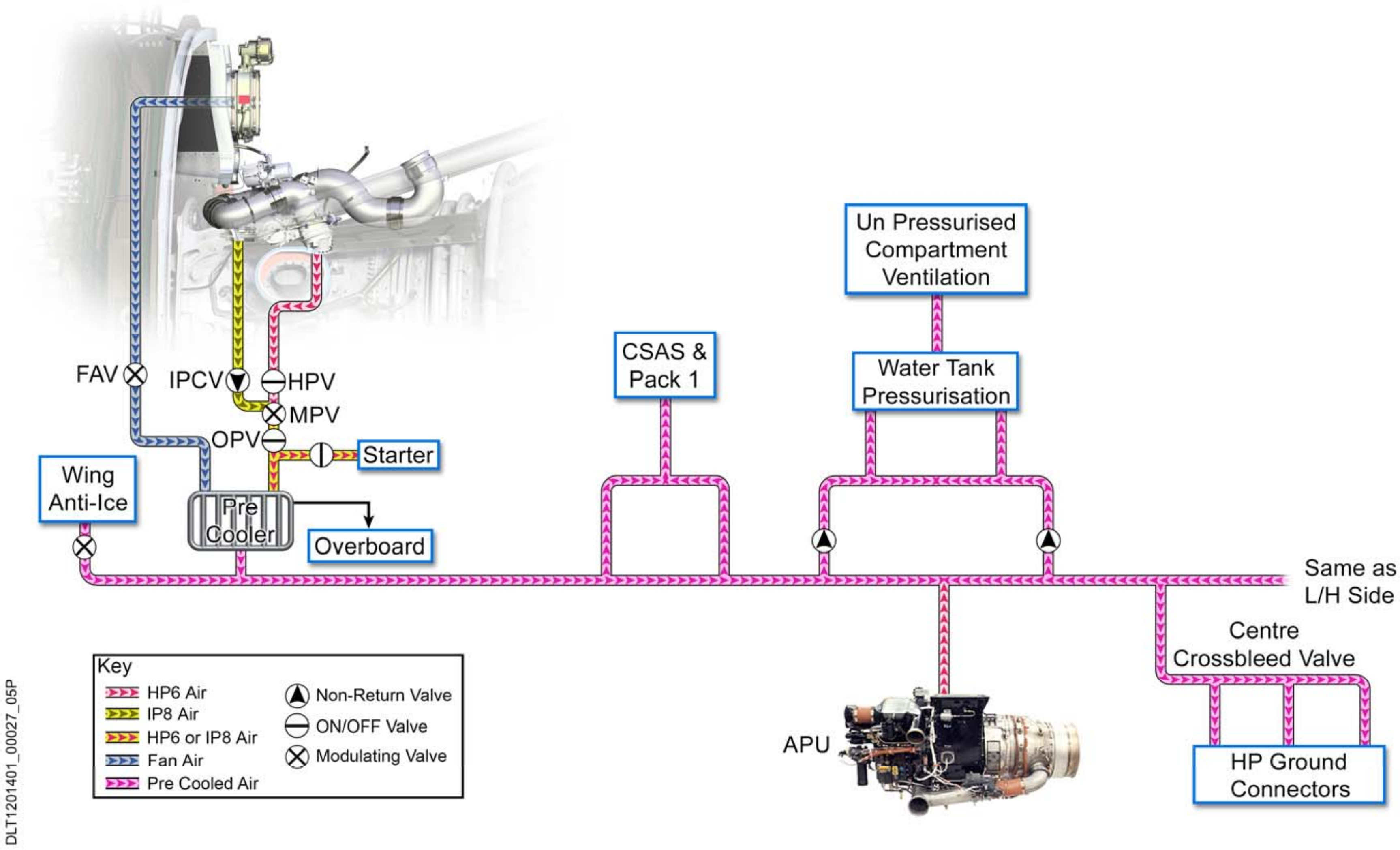
Interface

The EBAS interfaces with the Control and Display System (CDS) through the ADCN.

Control and Indicating

The main controls of the pneumatic system are the ENG 1 BLEED and ENG 2 BLEED pushbutton switches. They are on the AIR section of the overhead panel.

The main indications of the pneumatic system are on the BLEED page of the CDS.



FAV
IPCV
HPV
MPV
OPV
Starter
Wing Anti-Ice
Pre Cooler
Overboard
CSAS & Pack 1
Un Pressurised Compartment Ventilation
Water Tank Pressurisation
Same as L/H Side
Centre Crossbleed Valve
APU
HP Ground Connectors
Key
HP6 Air
IP8 Air
HP6 or IP8 Air
Fan Air
Pre Cooled Air
Non-Return Valve
ON/OFF Valve
Modulating Valve
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EBAS SYSTEM

Engine Bleed Air System (EBAS) Cont.

Engine Mounted LRUs

Location

The EBAS system is located on the engine between the thrust struts at the top of the core and within the pylon.

Purpose

The primary purpose of the EBAS system is to provide processed air into the cabin of the aircraft for crew and passenger comfort and life support. At low power settings HP6 air is supplying the air system and at higher power settings IP8 air is supplying the air system.

Description

The engine related LRUs of the EBAS system consists of the following components:

- IPCV – Intermediate Pressure Check Valve.
- HPV – High Pressure Valve.
- MPV – Manifold Pressure Valve.

Intermediate Pressure Check Valve (IPCV)

The IPCV is a split butterfly type valve located at the IP Bleed Port at the top of the intermediate case. The purpose of the IPCV is to:

1. Prevent high-pressure air (HP6) from entering the IP compressor when the HPV is open.
2. Allows IP compressor stage eight air to be supplied to the aircraft when the HPV is closed.

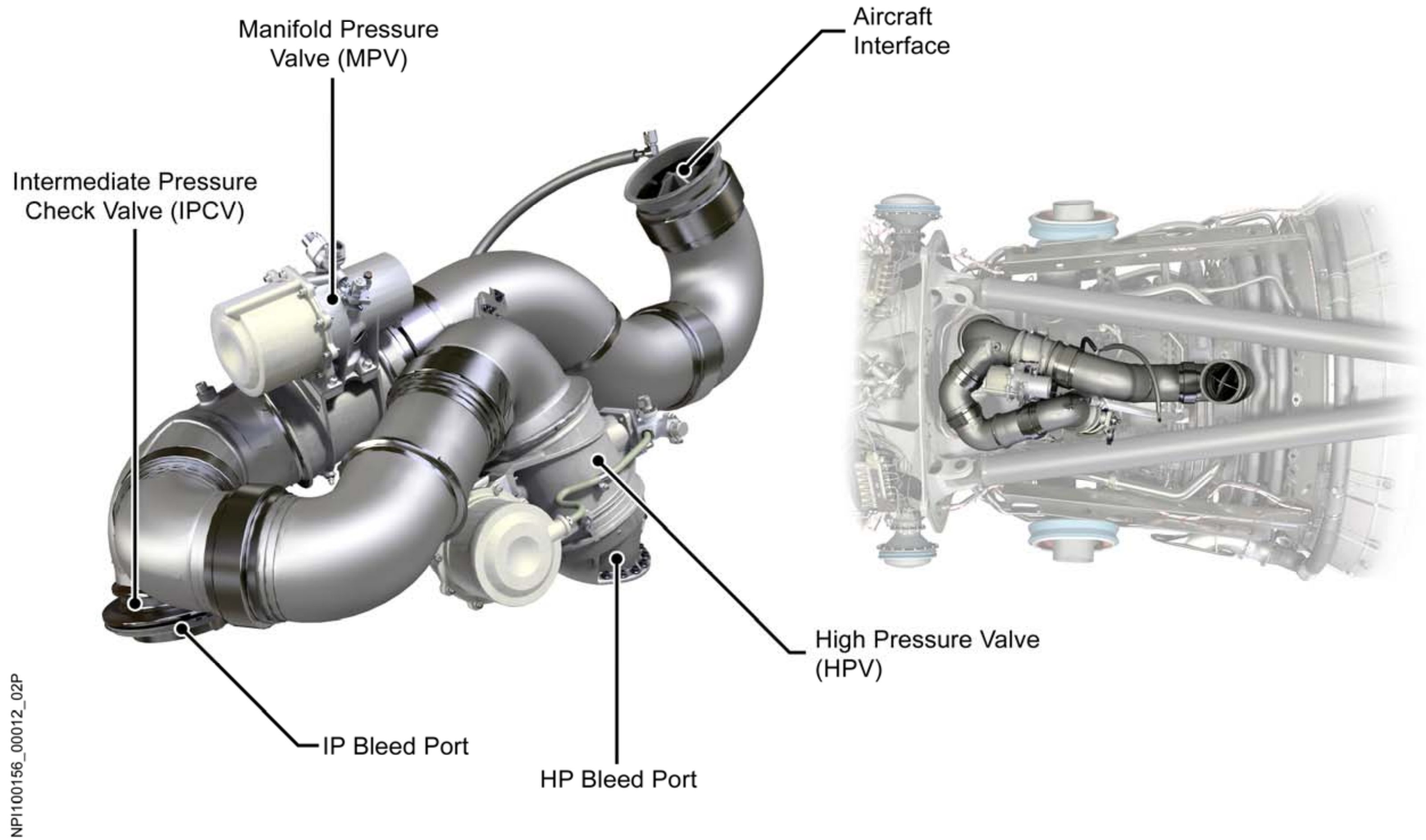
High Pressure Valve (HPV)

The HPV is located at the HP Bleed Port outlet at the top of the HP system. The purpose of the HPV is to control the flow of high-pressure air (HP6) entering the EBAS system.

The valve is controlled by the BAS application via the Environmental Control System (ECS) which is an electronic function within the aircraft systems. The EEC has no control of the HPV but can influence how long it is open or closed, based on the power being taken from the engine.

Manifold Pressure Valve (MPV)

The MPV is located in the duct prior to the aircraft interface. Its purpose is to regulate the pressure of the air entering the aircraft and is controlled by the BAS application.



ENGINE BLEED AIR SYSTEM (EBAS)

Engine Drains System

Location

The engine drains system is located on the lower half of the LP compressor case.

Purpose

The purpose of the engine drains system is to remove fluids that may leak from specific LRUs and areas of the engine, where fluids may create a fire hazard.

Description

The engine drains system consists of a number of tubes connected to certain LRUs and areas of the powerplant. The individual tubes come together at the combined breather and drains mast, to allow any fluids to drain overboard.

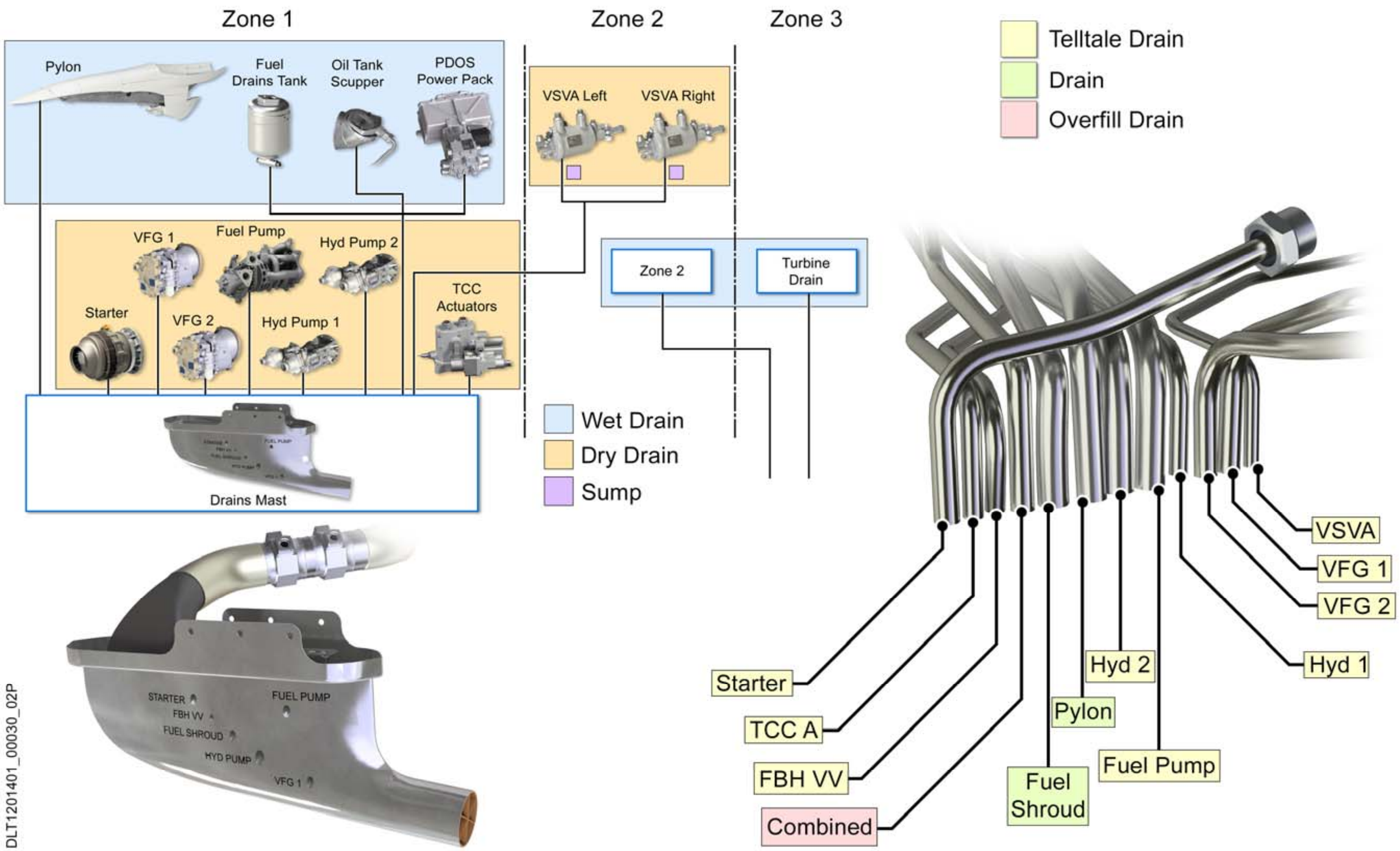
Each drain tube is considered to be either a 'dry' drain or 'wet' drain. Fluid is NOT expected to be seen from a dry drain. Fluid CAN BE expected from a wet drain. A limitation table contained within the AMP determines the acceptable level of leakage and subsequent action to be taken. The following is a list of the components or areas connected to the engine drains system and the fluid that may be seen.

Dry Drains

- Starter – oil.
- Variable Frequency Generator (VFG) front and rear – oil.
- Fuel pump – fuel or oil.
- Hydraulic pump yellow and green – oil or hydraulic fluid.
- Turbine case cooling valve actuators – fuel.
- Variable stator vane actuators – fuel.

Wet Drains

- Pylon – water, fuel, oil.
- Fuel drains tank, oil tank scupper and power door operating system power pack – This is a single combined drain where fuel or oil may be seen.
- Zone 2 – Water, fuel and oil.
- Turbine Drain – Fuel, water.



ENGINE DRAINS SYSTEM

Centrifugal Breather

Location

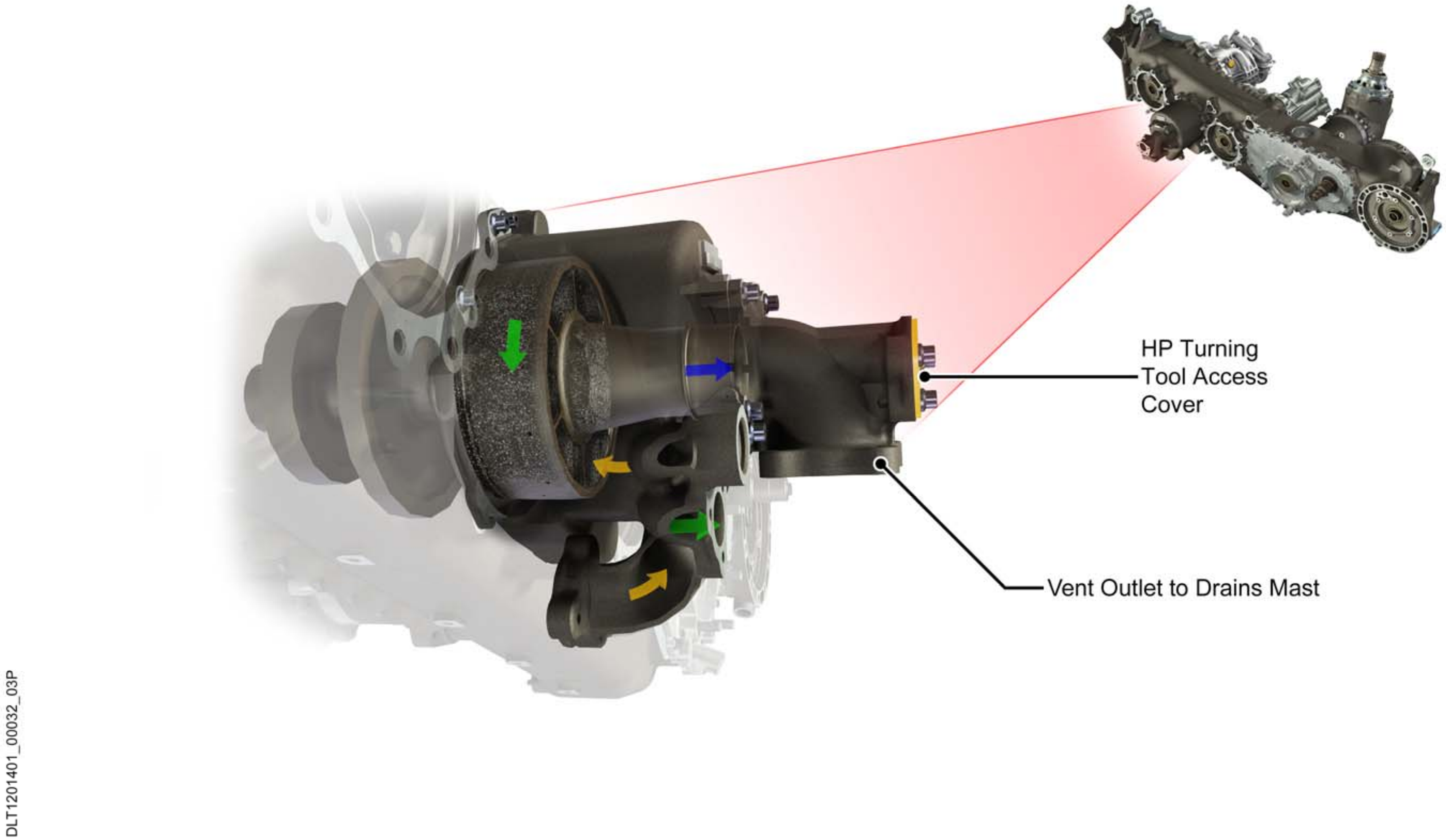
The centrifugal breather is located on the front right side of the external gearbox.

Purpose

Processes aerated oil from the bearing chambers vent system and oil tank. It separates the aerated mixture, the oil for reuse in the lubrication system and the waste air to be expelled overboard.

Description

The centrifugal breather has a rotor that contains retimet segments and is driven by the external gearbox. The delivered aerated oil is centrifuged out by the rotating action and scavenged back to the oil tank by a dedicated scavenge pump. The air passes through the retimet segments into the hollow rotor and is vented overboard through the drains mast outlet.



CENTRIFUGAL OIL BREATHER

Drains Tank assembly

Location

The fuel drains collector tank is located on the right side of the LP compressor case approximately the 4 o'clock position.

Purpose

The purpose of the drains tank is to collect the fuel remaining in the fuel manifold after a normal engine shutdown on the ground.

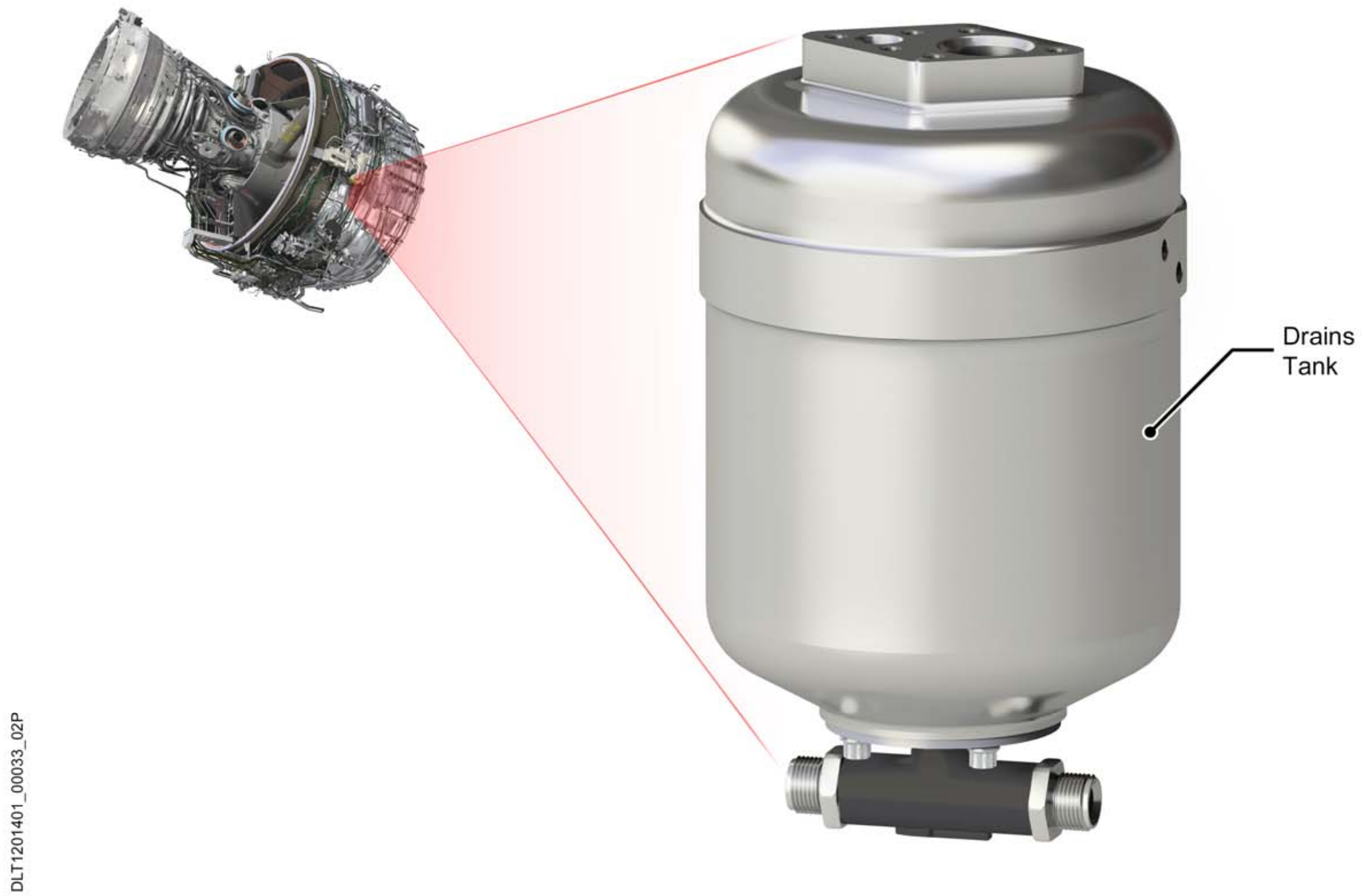
Description

The drains tank assembly consists of a steel tank and an ejector pump. Within the ejector pump is a non-return valve, a float valve and a filter assembly.

The drains tank collector is sized to hold fuel for one usual engine shutdown and three failed starts. If the number of failed starts is more than the maximum limit the drains tank can become full and the fuel will be dumped overboard through the drains mast.

Note:

The function of this component will be covered in detail within section 8 Fuel System.



DRAINS TANK AND EJECTOR PUMP

Hydraulic components

Location

Two pumps are located on the front face of the external gearbox, the Green system pump situated near the centre and the Yellow system pump off-set to the right, divided by the oil separator centrifugal breather.

Purpose

The purpose of the hydraulic pumps is to generate hydraulic pressure and flow to sustain the requirements of the aircraft and associated systems.

Description

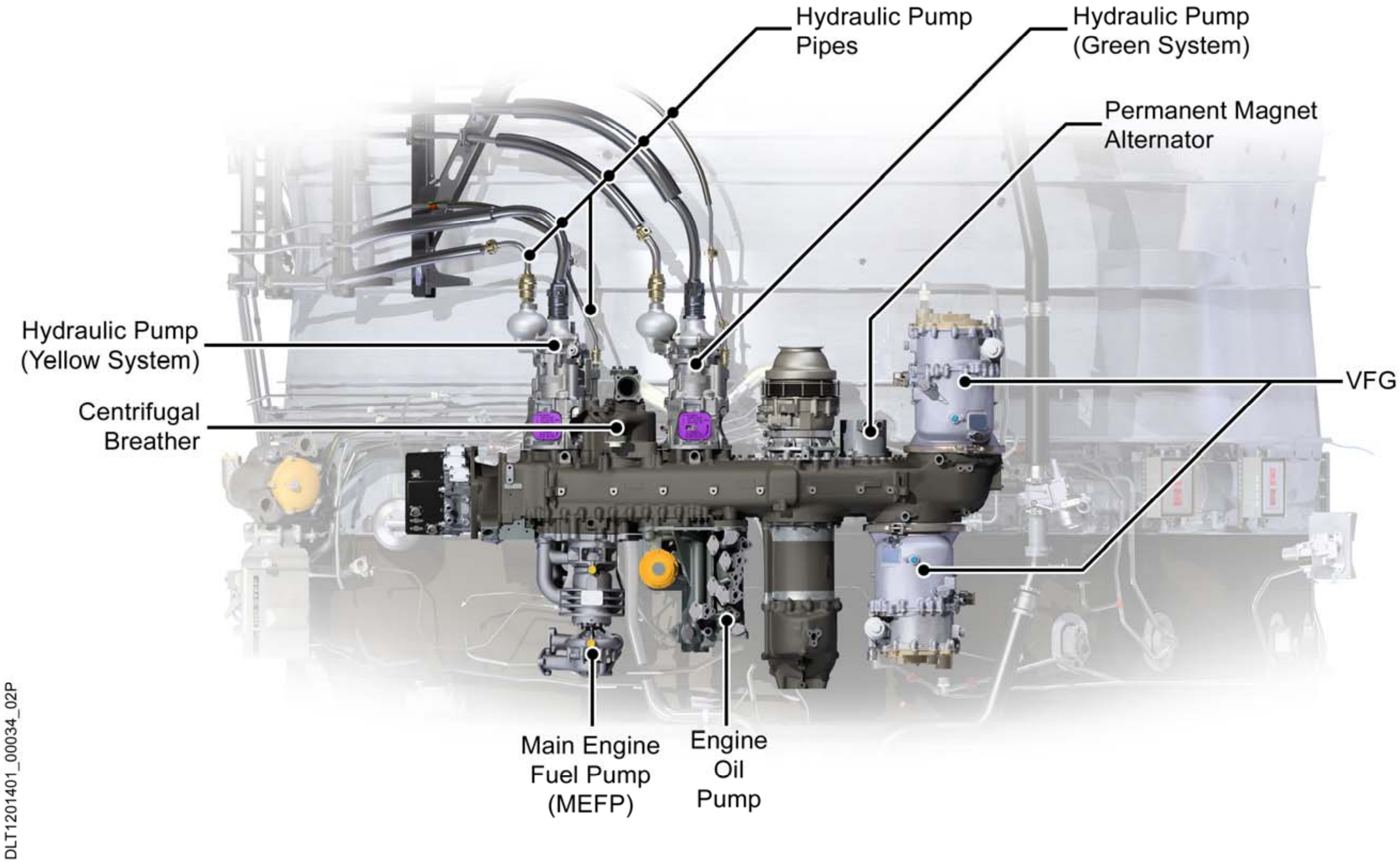
The function of the main hydraulic power system is to supply hydraulic power to the aircraft hydraulic systems. The green and yellow main hydraulic systems operate at the same time but are fully independently of each other and with no fluid transfer between them. The two Engine Driven Pumps (EDPs) for each system pressurise the hydraulic fluid to 5000 psi.

Each EDP provides pressurised hydraulic fluid for a main hydraulic power system. Each system has its own bootstrap-type hydraulic reservoir supplying hydraulic fluid to its two EDPs. The EDPs have a clutch to mechanically disconnect the pump drive shaft from the engine accessory gearbox; there is also a mechanical lockout for dispatch under MEL conditions. For cooling of the hydraulic fluid, heat exchangers are located in the wing tanks and are submerged in fuel; to act as cooling medium.

Definition

Bootstrap system: A Bootstrap system is a system that uses an external power source, but runs on an independent basis.

i.e. the engine provides hydraulic services to the airframe, but does not require hydraulic power for its own operation.



HYDRAULIC COMPONENT LOCATION

Electrical Generation Components

Location

The Variable Frequency Generators (VFGs) are fitted to the front and rear face of the engine external gearbox. The VFG Surface Air Cooled Oil Coolers (SACOCs) are fitted to the internal face of the engine fan case between the 6 o'clock and 9 o'clock position on the rear composite section.

Purpose

The VFGs provide the aircraft with electrical power, the SACOCs are used to cool the VFG oil system.

Description

There are two VFGs on the Trent XWB engine and each has a separated oil circuit / cooling arrangement.

Each oil circuit consists of the following components:

- A VFG oil return line temperature sensor.
- Heat exchanger matrix (SACOC)
- A de-congealing valve within the SACOC allows oil to bypass the SACOC matrix in the event of a blockage and during cold starting to improve SACOC de-congealing performance.
- An EEC controlled 2 position Thermal Bypass Valve (TBV).
- An anti-drain back check valve at the VFG inlet.

VFG Oil Temperature Sensor Location

- VFG oil temperature sensors are located at the VFG oil return lines.

VFG Oil Temperature Sensor Purpose

- To signal VFG oil inlet temperature to the EEC for monitoring and to control TBV operation.

EEC Controlled TBV Location

- In the oil bypass route between the SACOCs inlet and outlet.

EEC Controlled TBV Purpose

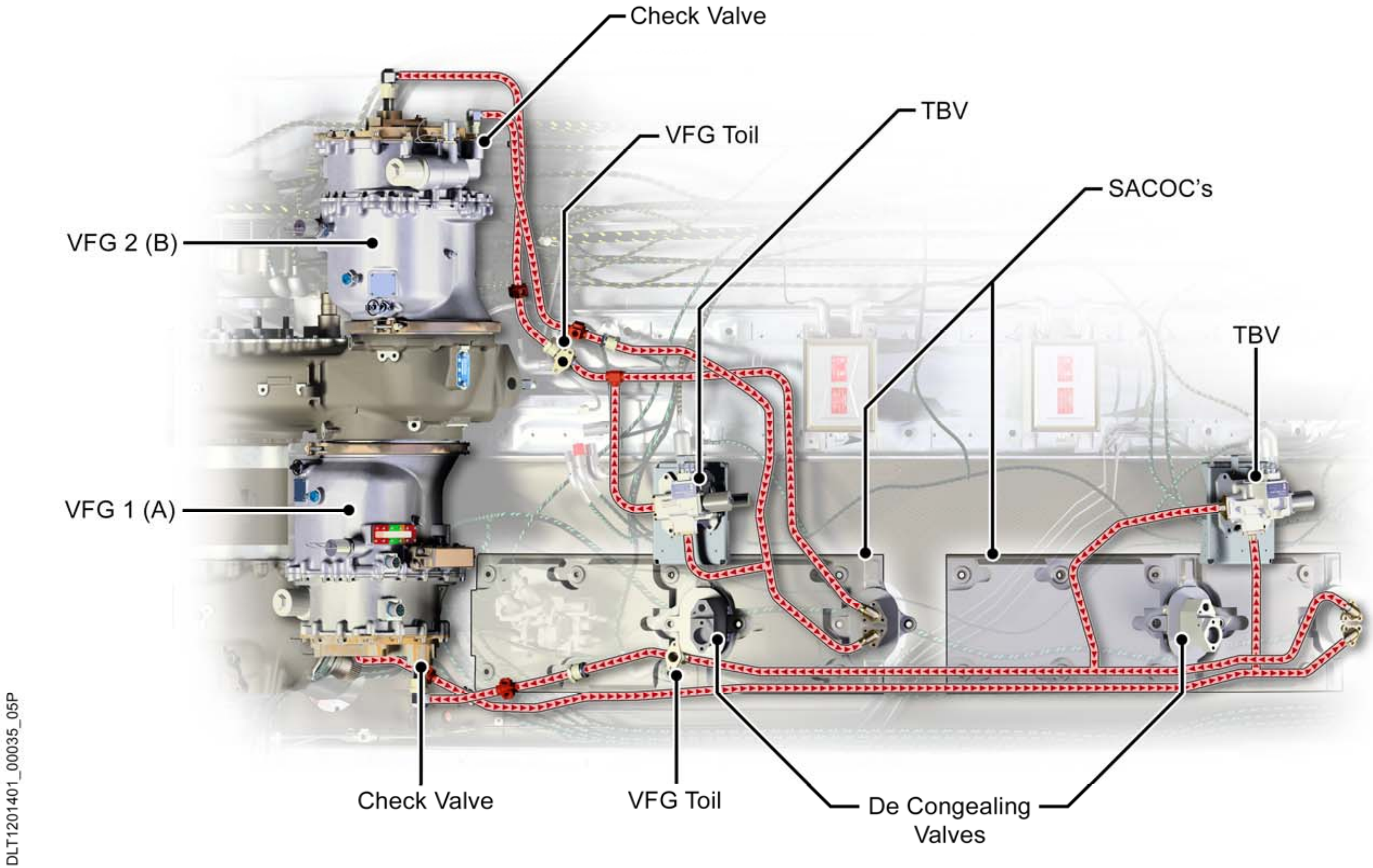
- Allows oil to bypass SACOCs matrix during cold operations to meet minimum oil temperature requirement based on VFG inlet oil temperature.

Check Valve

- Prevents oil draining back from the SACOCs when the engine is static

Function

The SACOCs uses Fan/Bypass air as the cooling medium, which follows continuously over its cooling fins while the engine is running, but the cooling oil when cold is bypassed. Increases in VFG oil temperature are sensed by the EEC, and then at a predetermined temperature the EEC signals the TBV to close, ensuring full oil flow through the SACOCs for maximum oil cooling.



VARIABLE FREQUENCY GENERATORS (VFGs) OIL SYSTEM

Variable Frequency Generators

Location

On the forward and aft face of the external gearbox on the left hand side.

Purpose

To generate electrical power at a rate of 230 VAC for use on the aircraft systems.

Description

There are two VFGs fitted to each engine, one on the forward and one aft face of the external gearbox, they are attached by quick release 'V' band clamps. The VFGs generate 230 VAC for the aircraft systems, and are cooled by their own oil system as previously described.

Operation

Normal Operation:-

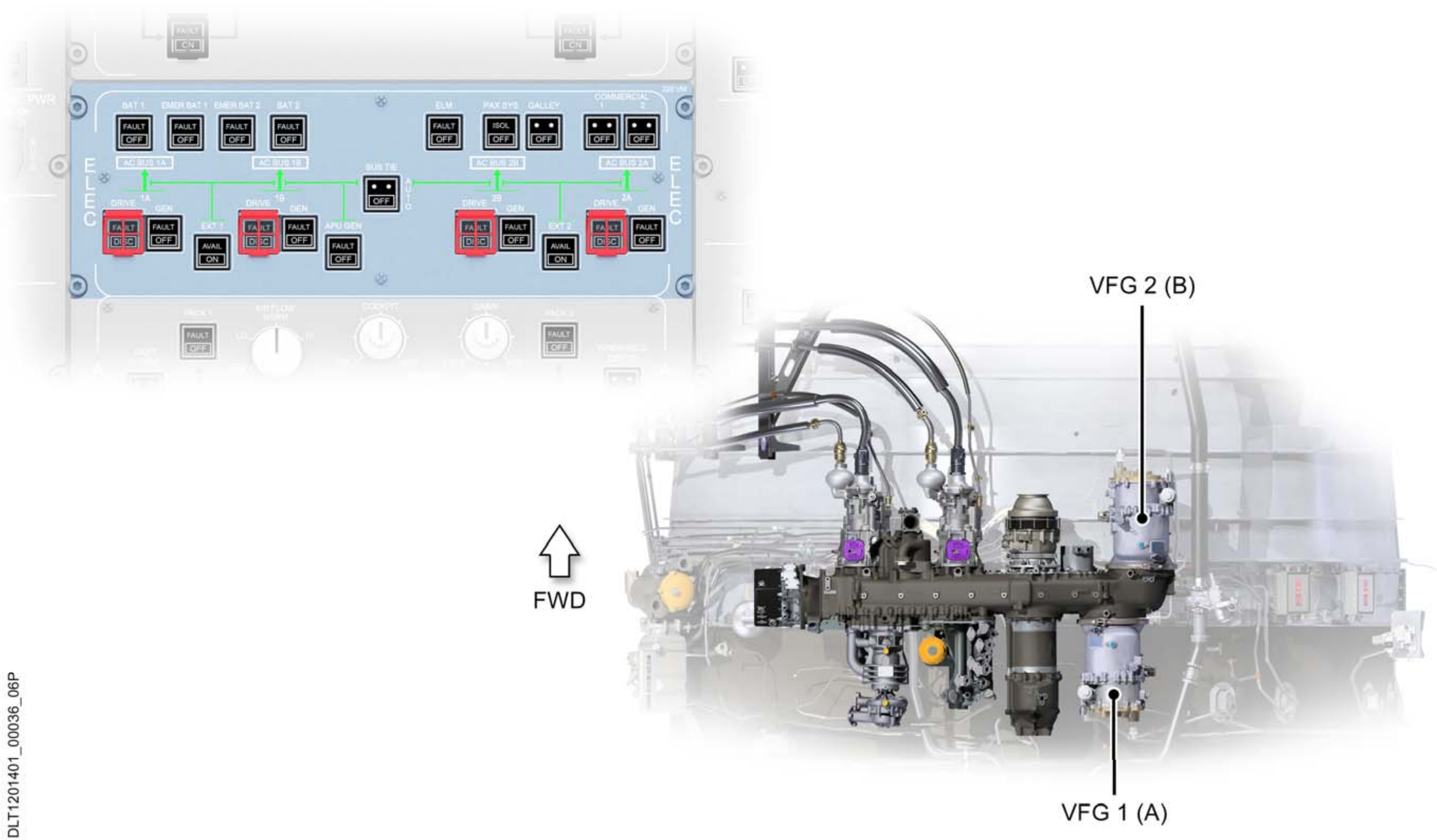
The GEN P/BSW is normally selected to the ON position; this allows the generator to come on line during engine start.

Note: Prior to the generators coming on line during engine starting, 'amber' GEN FAULT lights will be illuminated.

Manual disconnection:-
The DRIVE P/BSW controls the mechanical disconnection of the VFG.

If a mechanical defect of a VFG is detected, the 'amber' FAULT legend of the DRIVE P/BSW is illuminated. The cockpit crew must lift the safety guard and push the DRIVE P/BSW to energize the VFG-disconnect internal solenoid, which will illuminate the 'white' DISC legend in the DRIVE P/BSW.
The defective VFG rotor is then mechanically disconnected from the engine accessory gearbox.
If the troubleshooting is satisfactory, the maintenance personnel can re-connect the VFG to the gearbox with the related reset handle.

Automatic disconnection:-
The VFG internal disconnection solenoid operates when overheating occurs.
It is not possible to do a reset after a thermal disconnection: it is necessary to remove the defective VFG and send it to the applicable workshop.



VARIABLE FREQUENCY GENERATORS

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Variable Frequency Generator System

CAUTION

ANY SERVICING OR MAINTENANCE PROCESS MUST ALWAYS BE CARRIED OUT IAW WITH THE RELEVANT AUTHORISED MANUAL, AND NOT WITH ANY UNAUTHORISED DOCUMENTATION.

VFG Drive Disengage

During engine ground runs and when instructed to do so by the AMP the VFG can be disengaged from the gearbox drive remotely from the flight deck, this is done by a switch operation on the overhead panel.

CAUTION

VFG DRIVE DISENGAGEMENT IS NOT TO BE CONFUSED WITH THE GENERATOR OFF LINE ON LINE SWITCHES, THESE ARE SITUATED NEXT TO THE DRIVE DISCONNECT SWITCHES THAT ARE ON THE SAME ELECTRICAL PANEL.

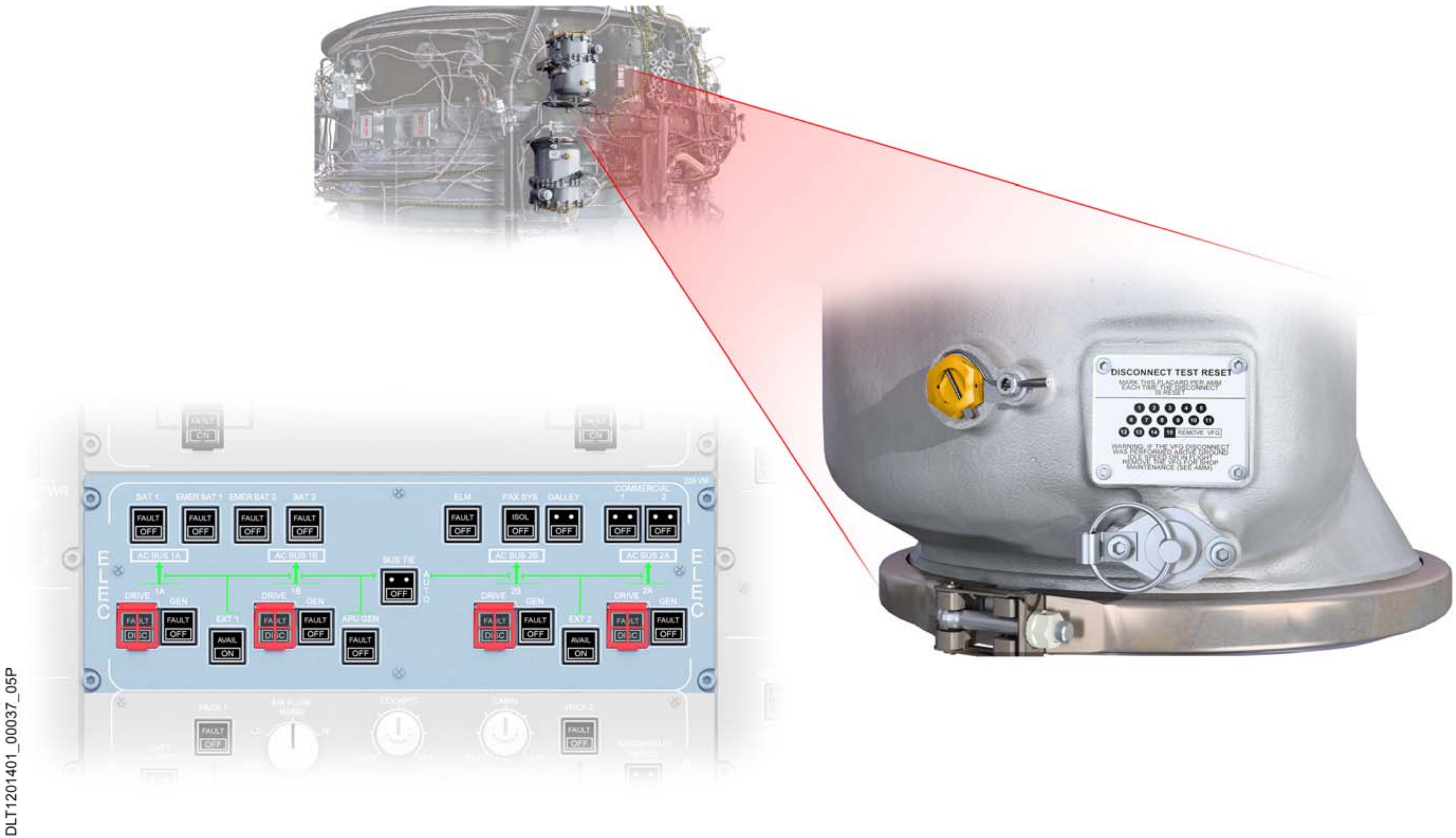
The generators can be selected on and off line multiple times as described in the AMP, but you are limited to the amount of times you can disconnect the drive from the VFG before it has to be removed for overhaul maintenance.

Drive Reset

The drive disconnect ring is secured into position with a ‘P’ clip, this is so that it cannot be accidently reset but it has to be a positive action to carry this task out. The reset must only be carried out once the engine is stopped.

CAUTION

IF THE DRIVE DISCONNECT IS CARRIED OUT ABOVE GROUND IDLE OR IN FLIGHT, THE APPROPRIATE AMP / TSM PROCESS MUST BE CARRIED OUT AND THE VFG REPLACED BEFORE FURTHER AIRCRAFT DISPATCH IS ALLOWED.



VFG DRIVE RESET

Maintenance Practices

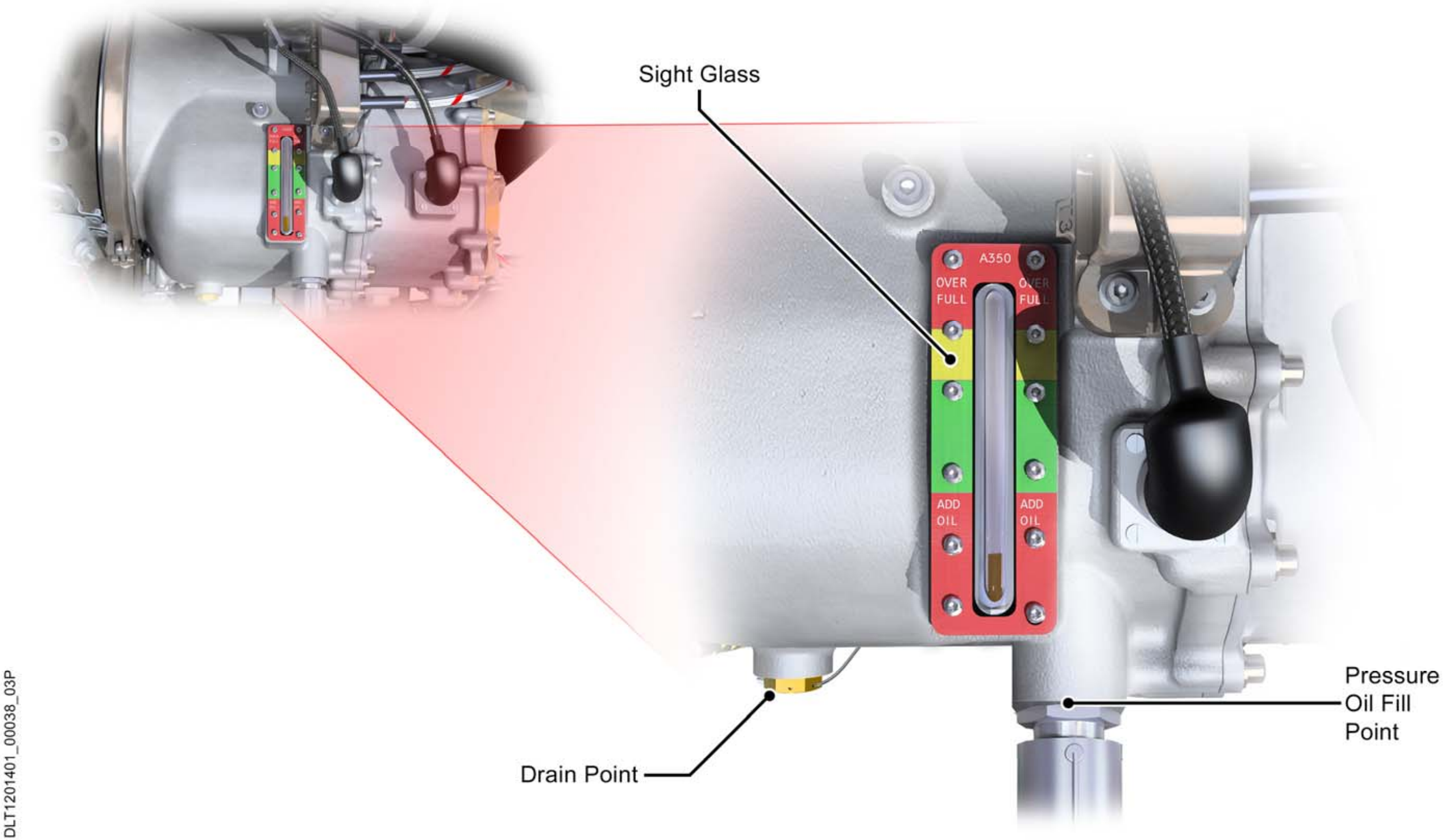
The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

VFG Oil servicing

As previously stated both VFGs have their own independent oil system, periodically these systems will require servicing.

This is achieved by each VFG having an independent pressure oil fill port and a sight glass. The sight glass graduations are calibrated in such a way that the relevant quantity indication is visible if the engine is fitted either to the right hand or left hand side of the aircraft. The pressure fill port is a standard attachment that is fitted to all Airbus aircraft.

Each VFG has its own integral oil filter and a drain line; the drain line is a dry drain therefore if oil is seen coming out of the drains mast the relevant troubleshooting should be carried out.



VARIABLE FREQUENCY GENERATOR SERVICING

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

VFG replacement

Warning: - waste oil is to be disposed of in an approved manner; harmful effects can be caused to you or the environment if this is not carried out iaw the relevant documentation.

The VFGs are a LRU, therefore a relatively easy and quick item to replace. This must be done IAW the relevant maintenance documentation.

Special tooling is required for the removal and refit of the VFG; this takes the form of a lifting trolley and an adaptor bracket.

All the other tooling is standard and should be easily available to the mechanic.

Prior to removal it will be necessary to drain down the oil from the VFG; this is accommodated with a sump drain port near the pressure fill point located on the VFG body.

Once replaced the VFG must be serviced and tested.

VFG SUPPORT JACK

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

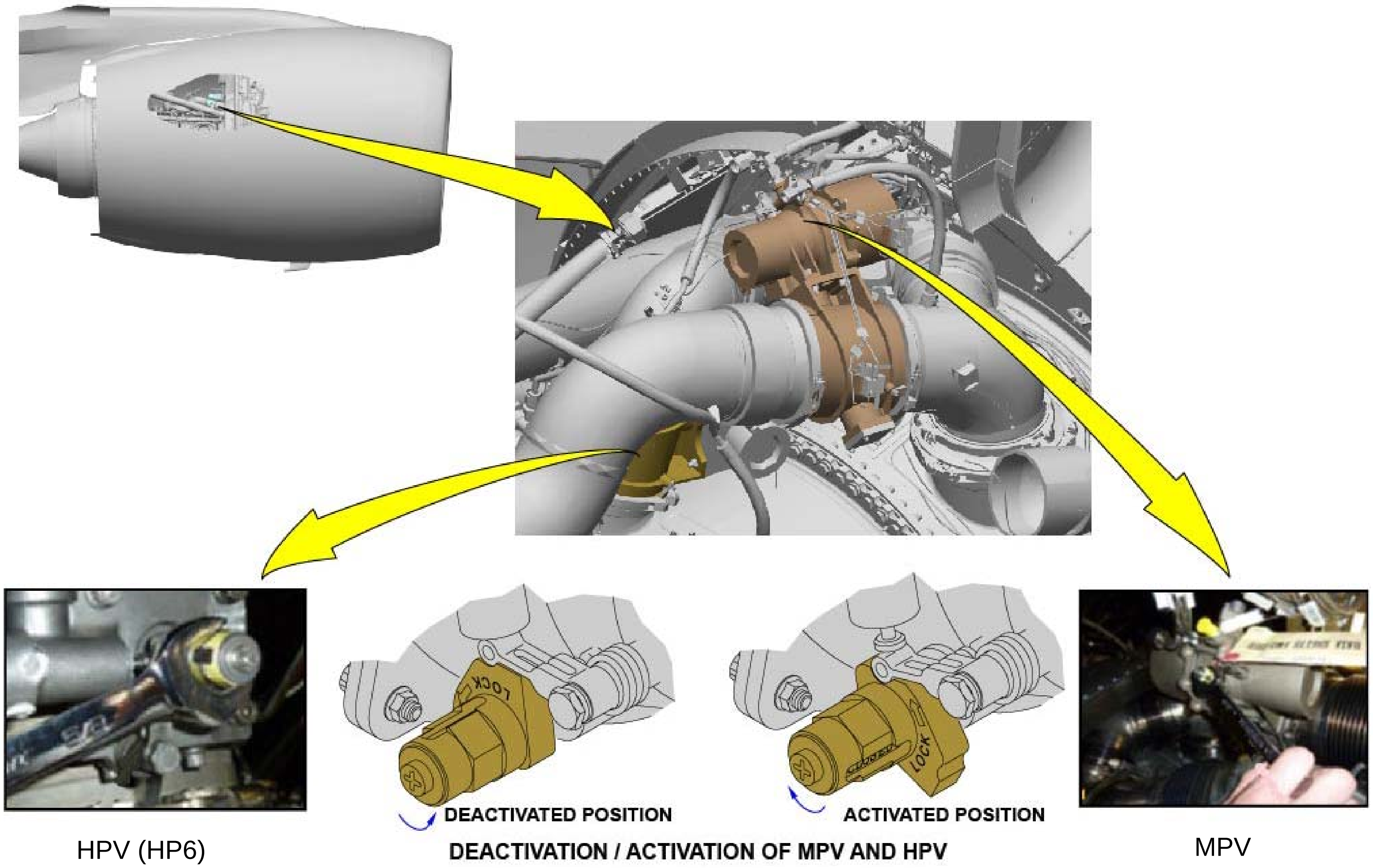
Engine Bleed-Air System (Deactivation)

The Manifold Pressure Valve (MPV) and High Pressure Valve (HPV) can be deactivated using a manual locking device with a visual position indicator.

The Fan Air Valve (FAV) can be deactivated using a manual locking device with a visual position indicator.

The Over Pressure Shut-Off Valve (OPSOV) valve can be deactivated through a quick access panel using a manual override and a locking screw.

The locking screw is also used as a visual position indicator.



EBAS DEACTIVATION

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Engine-Driven-Pump (EDP) Disengagement

For deactivation, a de-clutch mechanism is used to disconnect the EDP from the engine EGB. If you operate the declutch mechanism, the rotation/torque from the engine cannot be transmitted to the EDP. Thus, the EDP cannot turn and the supply of hydraulic power is stopped. This mechanism is a fully mechanical mechanism, which is manually operated on the ground.

To engage the EDP, again it is necessary to pull the release cable.

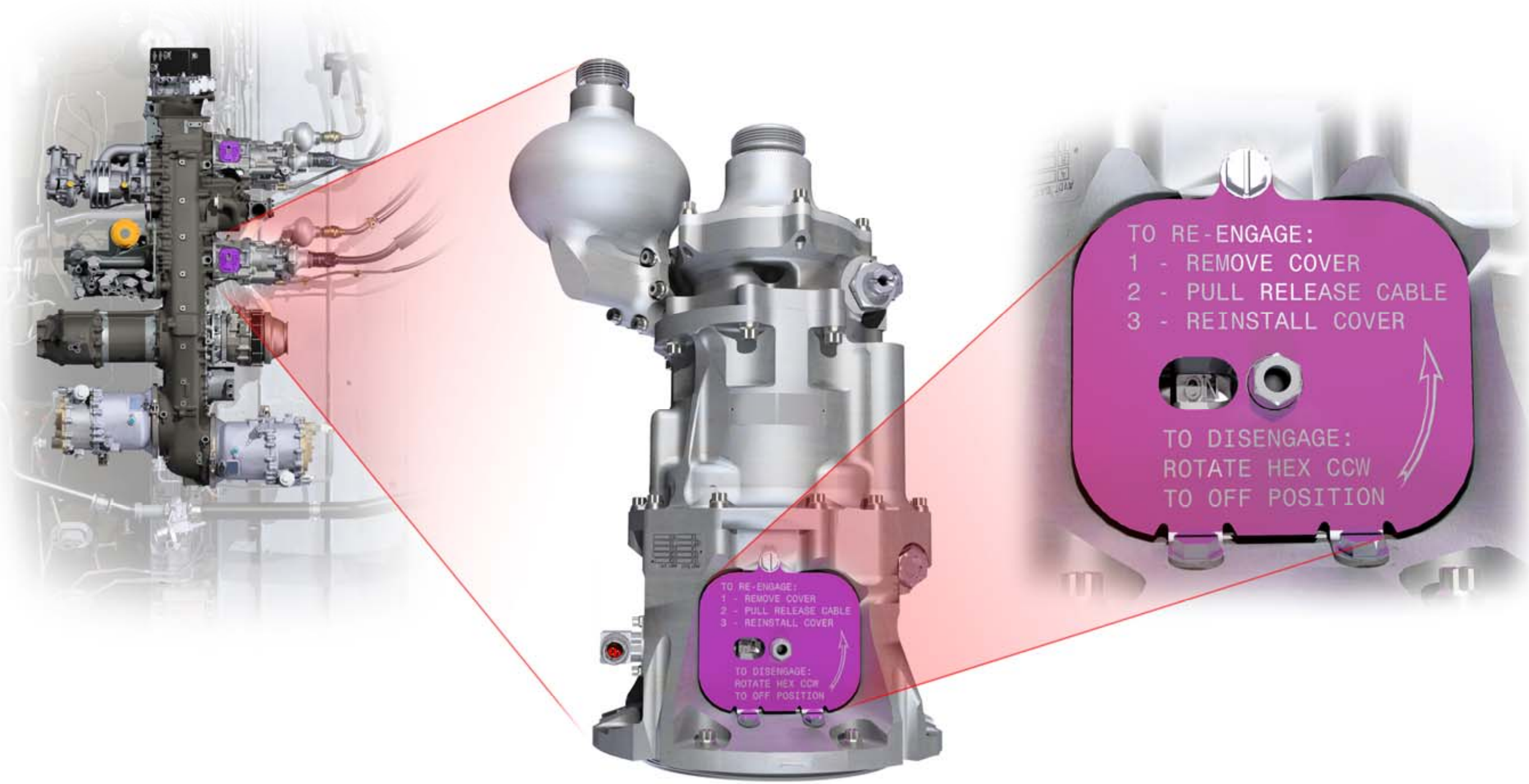
Hydraulic components

To manually disengage the Pump

1. Rotate deactivation feature counter clockwise
2. Make sure indicator shows ‘OFF’
3. Make sure disconnected (BITE test)

To manually engage the Pump

1. Remove screws and cover plate
2. Pull re-engagement ring
3. Make sure indicator shows ‘ON’
4. Make sure re-engaged (BITE check)
5. Install cover plate
6. Install and Torque 3 off screws



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HYDRAULIC PUMP LOCKOUT

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Hydraulic Components

Hydraulic Pump Replacement

Warning: - waste oil is to be disposed of in an approved manner; harmful effects can be caused to you or the environment if this is not carried out iaw the relevant documentation.

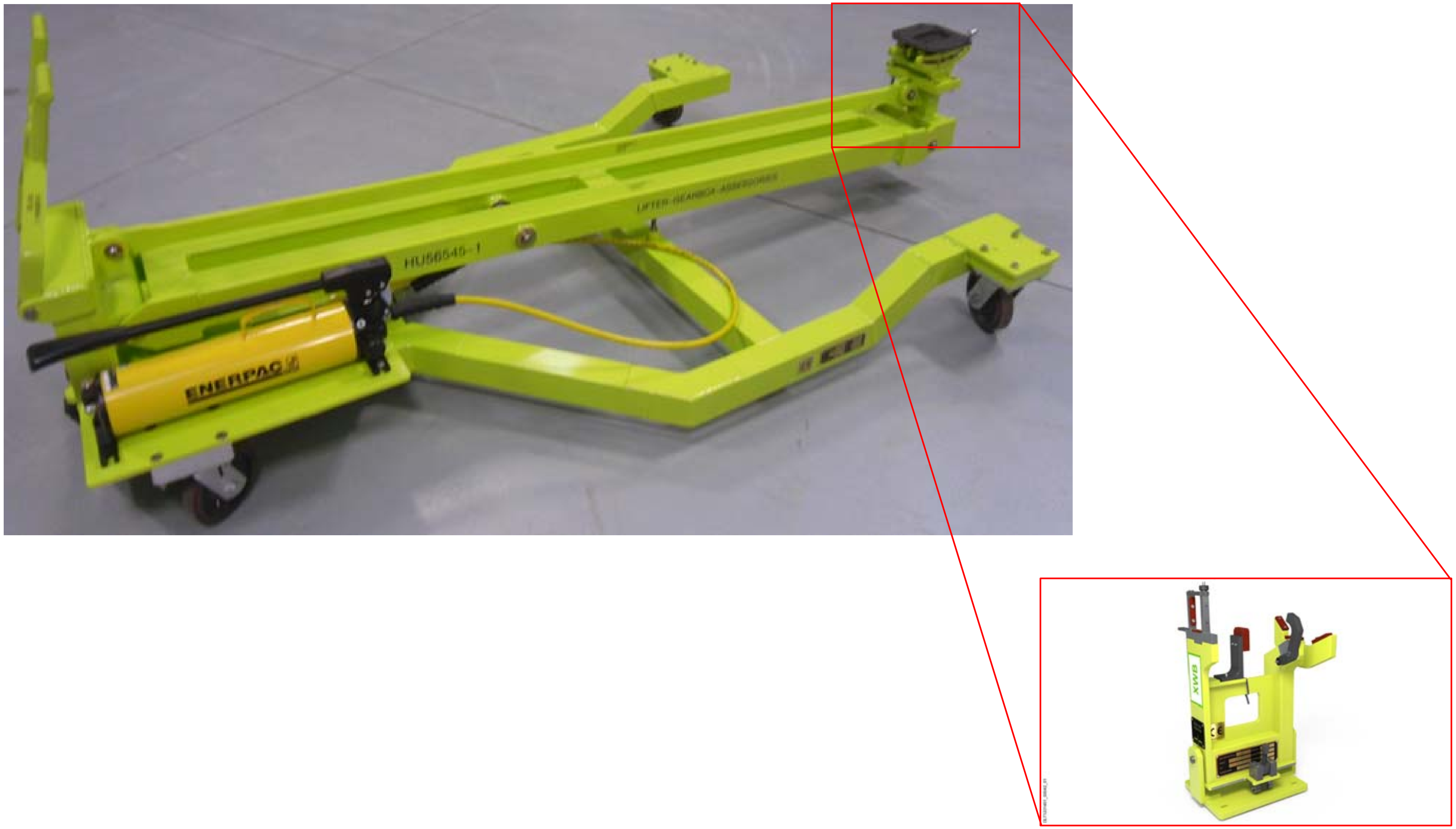
The Hydraulic pumps are LRUs, therefore a relatively easy and quick item to replace. This must be done IAW the relevant maintenance documentation.

Special tooling is required for the removal and refit of the Hydraulic pumps; this takes the form of a lifting trolley and hydraulic pump fixture.

All the other tooling is standard and should be easily available to the mechanic.

Prior to removal it will be necessary to drain down the oil from the Hydraulic pumps; this is accommodated with a sump drain port.

Once replaced the Hydraulic pumps must be tested IAW the relevant maintenance manual.

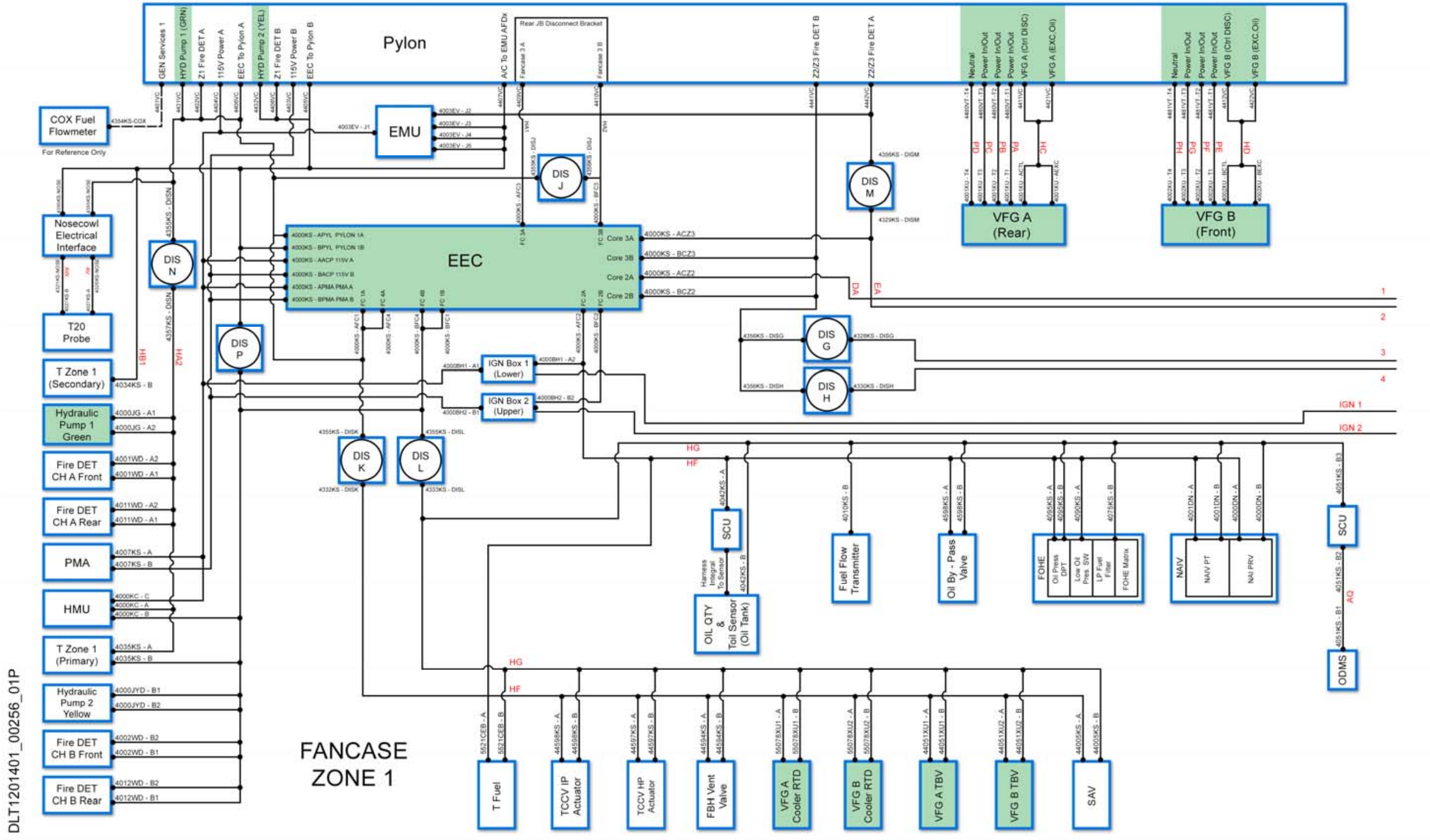


HYDRAULIC PUMP REMOVAL/INSTALLATION

Propulsion System Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.

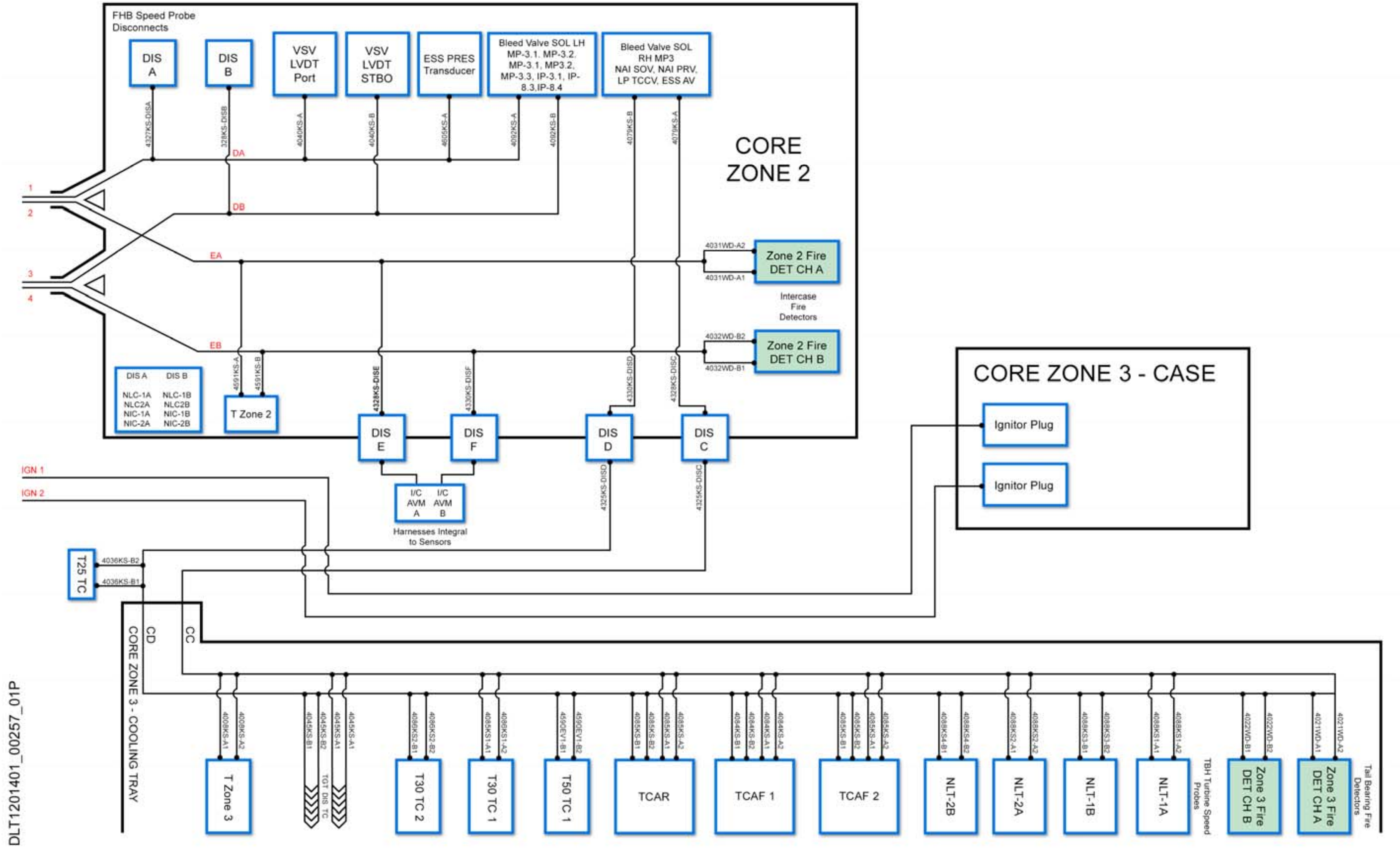


ZONE 1 WIRING DIAGRAM

Propulsion System Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.



ZONE 2 WIRING DIAGRAM

Section 3 – Propulsion System

Objectives

At the end of this section the student will be able to:

- State the purpose of the propulsion system.
- Locate and identify the major assemblies that form the propulsion system.
- Describe the purpose and operation of the major assemblies that form the propulsion system.
- State the WARNINGS & CAUTIONS associated with the Propulsion system.
- Describe the Engine Bleed Air System (EBAS) components installed on the engine and their main purpose.
- Describe the Engine Drains Mast.
- Describe the Engine Hydraulic components installed on the engine and their main purpose.
- Describe the Engine Electrical Generation System components installed on the engine and their main purpose.
- Describe the Engine Oil Breather.

End of Propulsion System Section

Section 4 – Mechanical Arrangement

Section 4 - Mechanical Arrangement

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance documentation level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the bearing arrangement of the Trent XWB engine.
- Recognise the modular breakdown of the Trent XWB engine.
- Identify the location and describe the purpose and operation of the engine modules.
- Identify and locate the Trent XWB engine components installed on the left and right side of the engine.
- Identify the location and describe the purpose of the borescope access positions on the Trent XWB engine.

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Main Rotating Assemblies

Introduction

The engine utilises three main rotating assemblies for efficient thrust production.

Location

The three rotating systems form the main assembly of the engine and are located throughout the engines internal area.

Purpose

The purpose of the rotating assemblies is to compress all the air entering the engine. The compressed air is used primarily to produce thrust from the Fan and produce a high energy gas flow when burnt with fuel in the combustion chamber. The High Pressure (HP) rotating system also drives the external gearbox which in turn drives the mounted accessory components and so allowing the engine and aircraft systems to be operated.

Description

The three rotating assemblies are called the:-

- Low Pressure (LP),
- Intermediate Pressure (IP)
- High Pressure (HP) systems

Each system is mechanically independent from the other but each system can affect the other two.

The rotating assemblies operate separately to each other with the HP system on the outside, the IP system in the middle, and the LP system in the centre. Each system has a compressor, a shaft and a turbine.

Burning of the fuel / air mixture in the combustion chamber produces a gas flow with increased volume and kinetic energy; this energy is used by the turbine to create a rotation force which drives the shaft that connects the turbine to the compressor.

Low Pressure System

The Low Pressure (LP) system rotates counter clockwise and comprises of a single stage LP Compressor (Fan) connected by a shaft to a six stage LP Turbine.

Intermediate System

The Intermediate (IP) system rotates counter clockwise and comprises of an eight-stage compressor connected by a shaft to a two stage IP turbine.

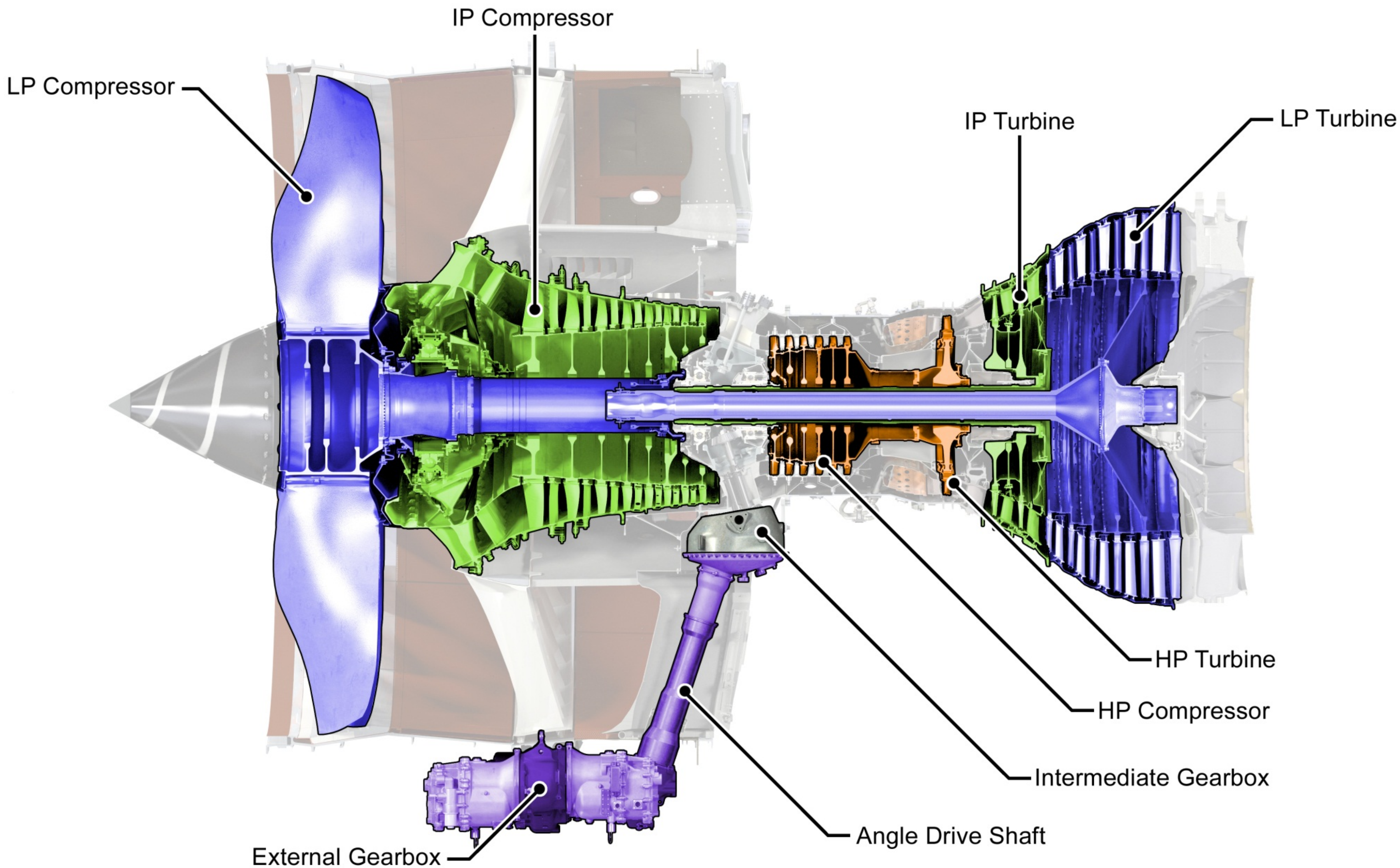
High Pressure System

The High Pressure (HP) system rotates clockwise and comprises of a six-stage compressor connected by a shaft to a single stage turbine.

Each of the rotating assemblies is supported independently by a combination of roller (support) bearings and ball (location / thrust) bearings.

The External Gearbox (EGB) is driven from the HP system via the Internal Gearbox (IGB) and the Intermediate Gearbox.

IGB bevel gears transmit HP shaft rotation force to the intermediate gearbox bevel gears, and then via the radial drive shaft the rotation force is transmitted to drive EGB.



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ENGINE MAIN ROTATING ASSEMBLIES

Engine Main Bearing Arrangement

Introduction

Bearings are used within engineering assemblies to allow rotational movement between two parts i.e. a shaft and a structure.

Location

The bearings are located inside the engine in four main areas. These areas are:

- Front Bearing Housing (FBH)
- Internal Gearbox (IGB)
- HP / IP Bearing Chamber
- Tail Bearing Housing (TBH)

Purpose

Bearings provide a means of accurately supporting and locating the rotors whilst transmitting force and offering minimal rotational resistance.

Description

Two types of bearings are used in the Trent XWB engine, ball bearings and roller bearings.

Ball (location) bearings are located in the Internal Gearbox (IGB) of the Compressor Intermediate module and the Front Bearing Housing (FBH) of the IP Compressor module.

Ball bearings can withstand radial and axial forces and are therefore suitable for transmitting thrust and locating shafts.

Roller bearings are located in all the main areas mentioned above and transmit radial loads while allowing axial movement of the shaft.

The LP and IP rotor assemblies are each supported by three bearings. The HP rotor is supported by two bearings.

LP Rotor Assembly

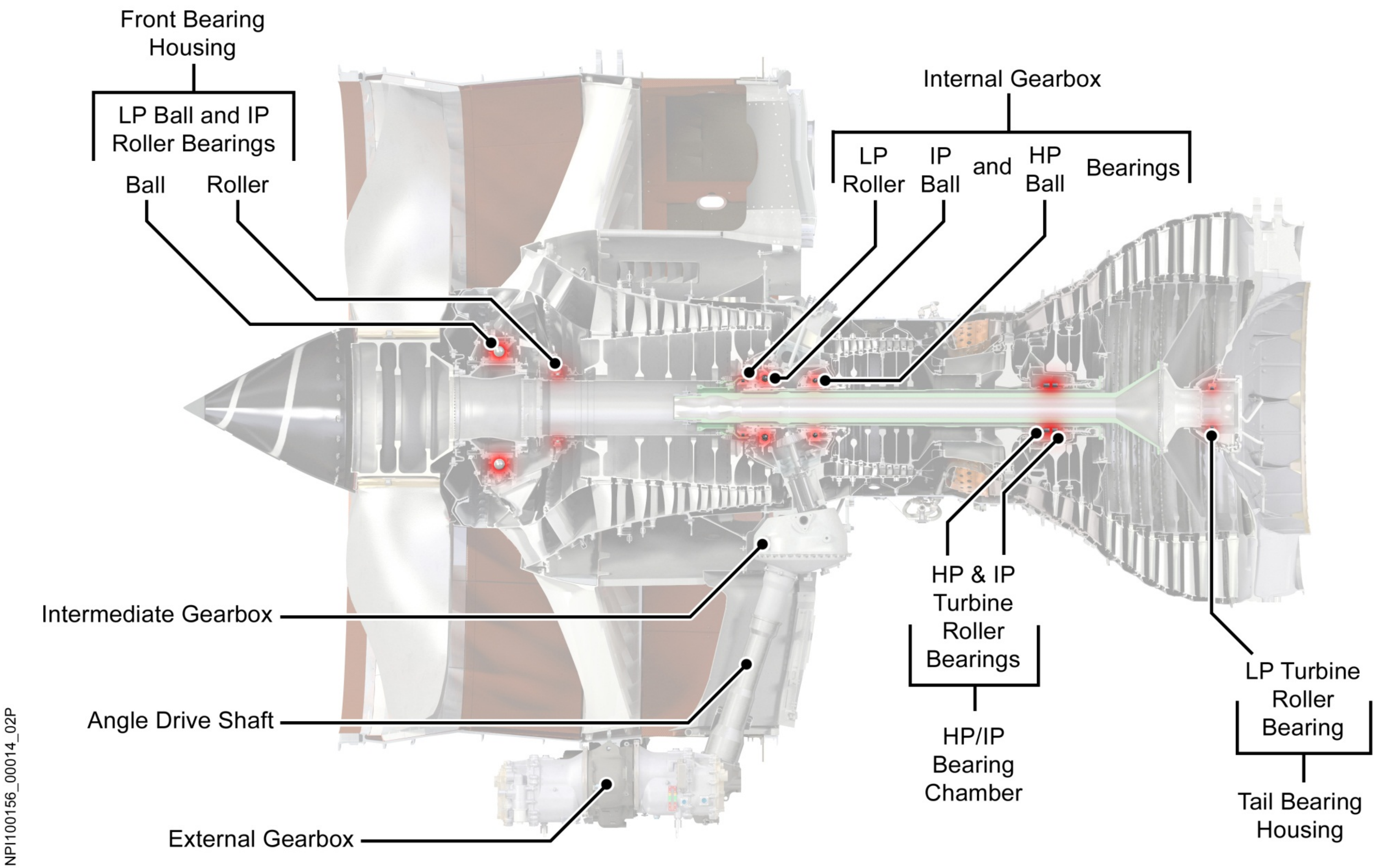
LP rotor system is supported by two roller bearings and a single ball (location) bearing. The LP ball bearing in the FBH positions the LP compressor shaft for location, support and thrust transmission to the rear mount. The central roller bearing in the internal gearbox supports the rear of the LP compressor shaft and the front of the LP turbine shaft. The final roller bearing in the TBH supports the rear of the LP turbine shaft.

IP Rotor Assembly

The IP rotor system is supported by two roller bearings and a single ball bearing. The front roller bearing supports the IP compressor and the rear roller bearing supports the IP turbine. The central ball (location) bearing positions the shaft and also transmits the thrust developed by the IP system.

HP Rotor Assembly

The HP rotor system is supported in two positions. A roller bearing supports the HP turbine. The ball (location) bearing supports the HP compressor, locates the HP system, and also transmits the thrust developed by the HP rotor system.



ENGINE MAIN BEARING ARRANGEMENT

Reasons for moving the XWB LP Location bearing from the Internal Gearbox to the Front Bearing Housing

- The loads imposed on the Trent XWB turbine are greater than that of previous Trent's.
- Larger bearings required more room, but not enough room in IGB.
- FBH large enough to take 52mm ball (Thrust) bearing.
- By increasing the size of the LP location bearing the amount of air into the IP Drum can be reduced.
- IP Drum traditionally acts as a piston to counter act the effect of the turbine. In these traditional design large quantities of IP5 air is directed into the IP Drum to pressurize the 'piston'.
- Fan air seal leakage adds to inefficiency.
- By moving the LP location bearing to the FBH not as much IP5 air is required therefore saving fuel.

Front Bearing Housing

Location

The front bearing housing is located on the forward end of the Intermediate compressor module.

Purpose

The front bearing housing has several purposes:

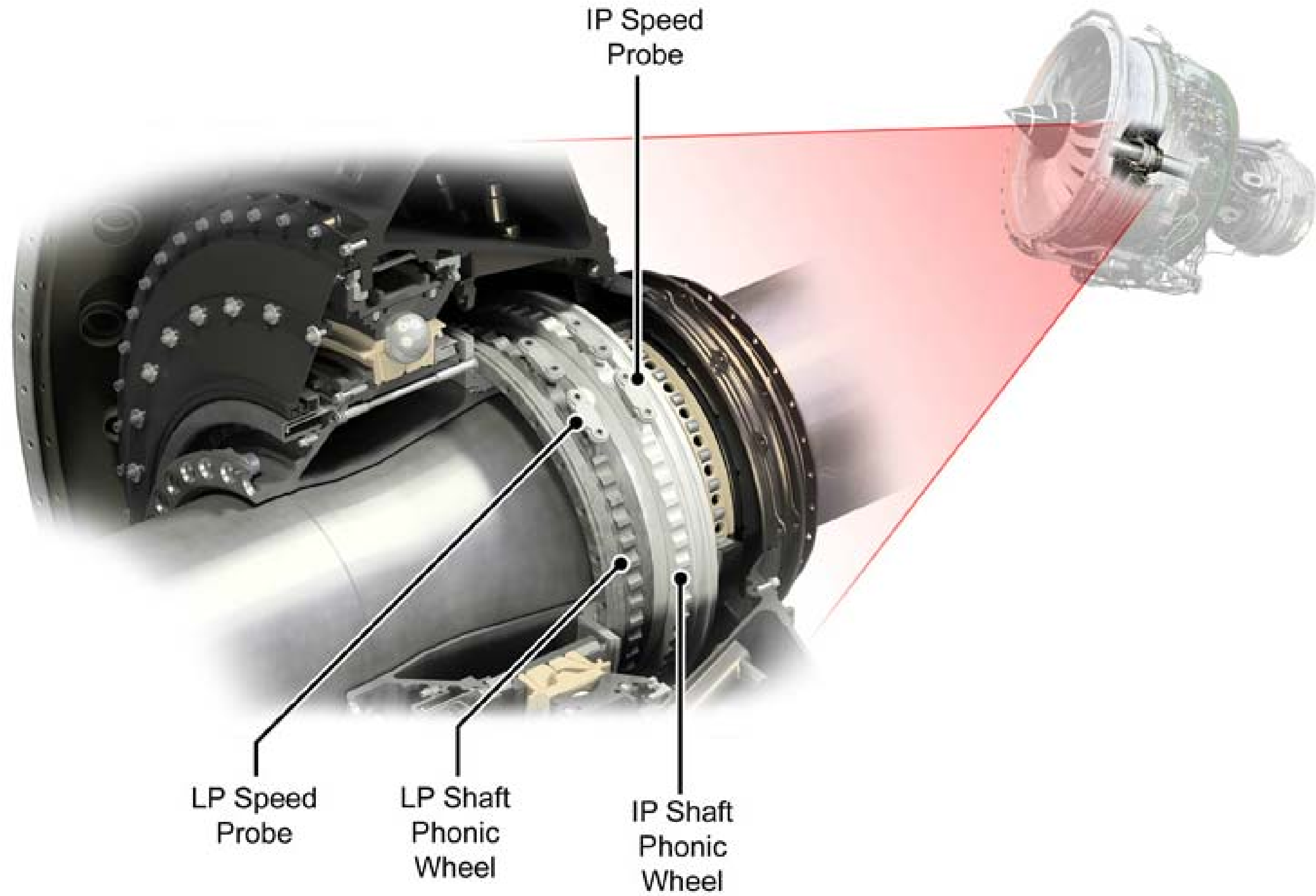
- To provide a sealed chamber to house both the LP thrust bearing and forward IP roller bearing.
- House both the LP and IP Phonic wheels and Speed Probes.
- Provide lubrication of the bearings.
- Provide a structure to support and transmit thrust loads.

Description

The inner section of FBH is formed by both LP and IP systems rotating drive shafts. The outer section is formed by static sections which are bolted to the Fan exit guide vane inner ring.

The bearings within the housing are lubricated using oil jets and squeeze film lubrication, this flow of oil pressurises the housing, which has to be sealed to allow the oil to be recycled and prevent the oil from escaping into the engine main air flow.

The FBH utilises carbon seals supported by IP stage 5 air to seal the forward and rear sections of the bearing housing. Within the housing where the LP and IP shafts interact, sealing is achieved by using a hydraulic type seal.



FRONT BEARING HOUSING

Trent XWB Modular Breakdown

Introduction

The Trent XWB is designed as a series of modules to ease assembly and overhaul. A module consists of a number of parts in a self-contained unit that interface with other modules and systems.

Purpose

The purpose of building the Trent XWB Engine in a modular format is to ease manufacture, engine assembly, maintenance and repair.

Description

There are seven modules, that when assembled together, are connected to the associated engine systems to form the Trent XWB engine. Each module is numbered in accordance with the Air Transport Association (ATA / S1000D) numbering convention. This type of construction uses a modular approach that offers a number of important benefits such as:

- Decreasing repair turn-round time.
- Lowering overall maintenance costs.
- Reducing spare engine holdings.
- Maximizing module life.
- Easier transportation and storage.
- More economical transportation and storage.

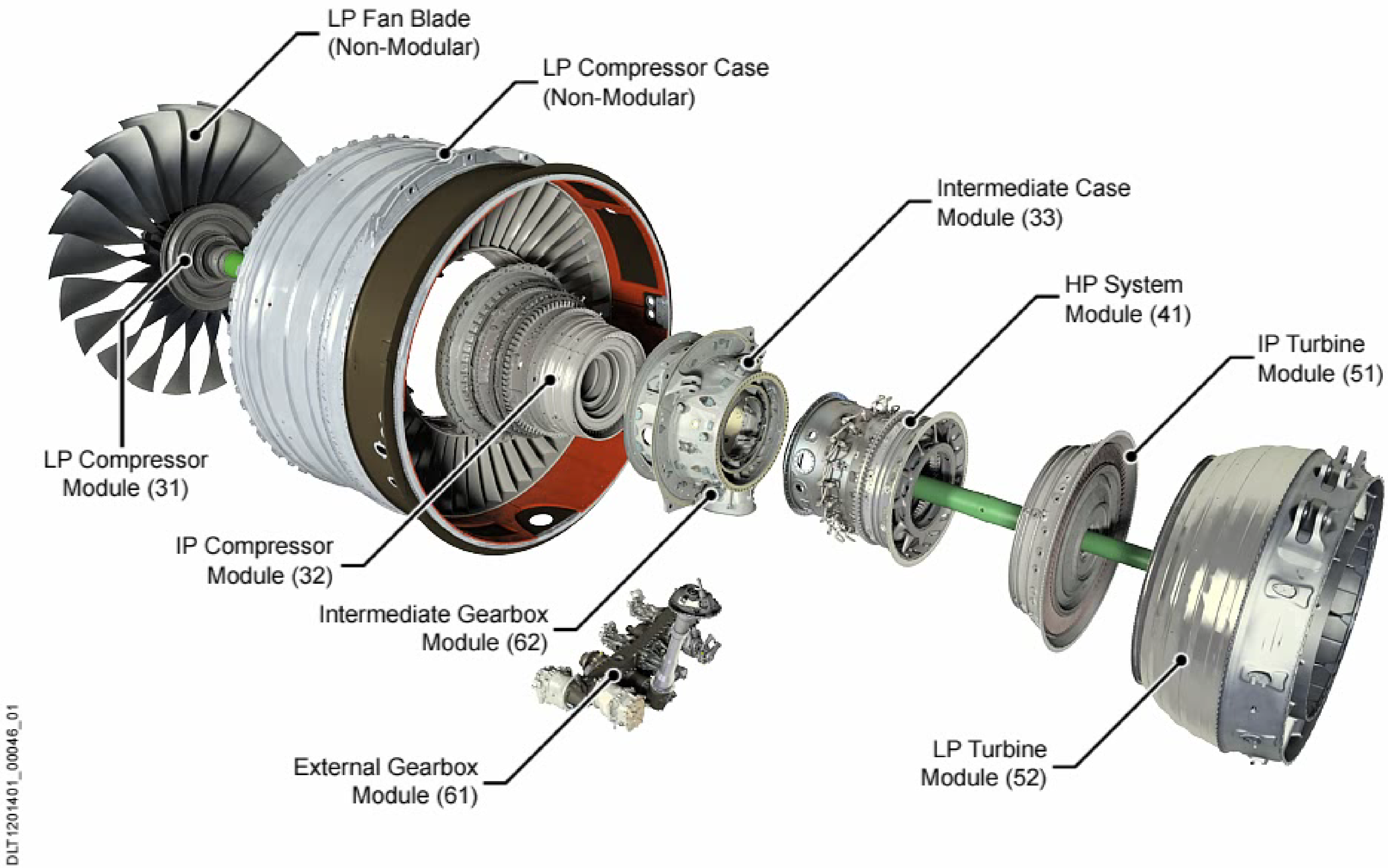
The Trent XWB engine seven modules are numbered and named as follows:

- Module 31 - LP Compressor Rotor Module.
- Module 32 - IP Compressor Module.
- Module 33 - Compressor Intermediate Module.
- Module 41 - HP System Module
- Module 51 - IP Turbine Module.
- Module 52 - LP Turbine Module.
- Module 61 - External Gearbox Module.

The modules are connected to their respective partners i.e. compressor to turbine, by an axial drive shaft and to the neighbouring module by bolts on the outer flange. The external gearbox is mounted to the bottom of the LP Compressor Case and driven by a radial shaft connected to the intermediate gearbox.

Non-Modular Components

Components that are not part of the module are described as being non-modular such as tubes, harnesses and in the case of the Trent XWB the LP Compressor Case and the LP Compressor Blades. The annulus fillers and spinner assembly are also classed as non-modular items.



TRENT XWB MODULAR BREAKDOWN

LP Compressor Module

Location

The LP Compressor is located at the front of the engine.

Purpose

The purpose of the LP compressor module is to provide location and drive to the LP compressor blades.

Description

The LP compressor module consists of the fan disc, LP compressor shaft and fan air seal. Together they are called the module 31.

Fan Disc

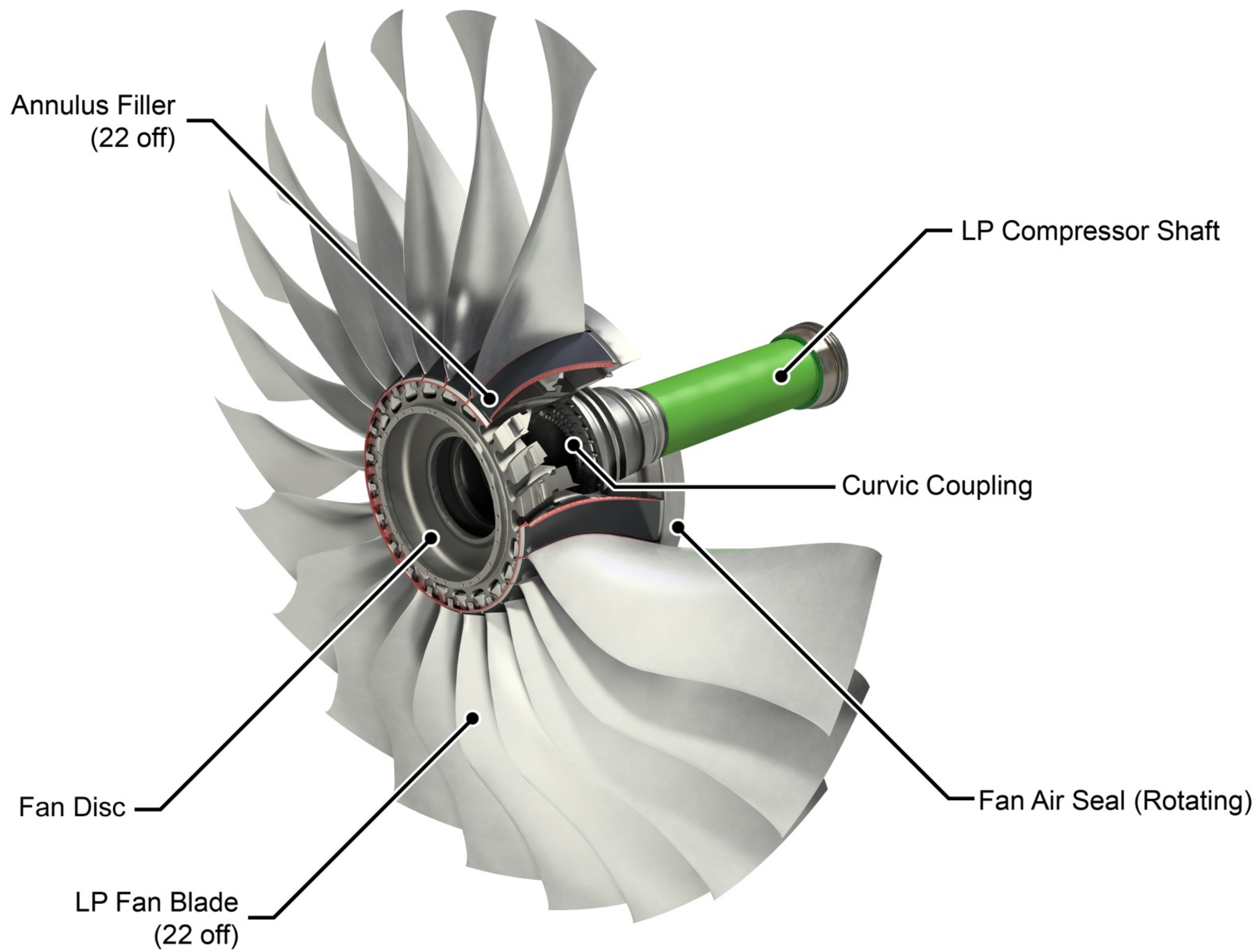
The fan disc retains the fan blades in position using 22 curved axial dovetail profile that react to the loads from the blades. A single radial cut out is machined into each dovetail profile to locate a shear key at the bottom of the fan blades to secure the blades axially. The number 1 blade is identified by ←**1** engraved on the disc face.
Annulus fillers are installed between the blades to give a smooth contour to the internal surface and provide the inner annulus of the LP compressor.

LP Compressor Shaft

The LP compressor shaft is bolted to the fan disc through a double row curvic coupling. Behind the coupling is the LP location bearing, which keeps the shaft in the correct radial position. At the rear it is attached to the LP turbine with a splined coupling. On the shaft is a phonic wheel that interacts with static speed sensors to relay LP rotational shaft speed (N1). LP system trim balancing is obtained by using a single shorter (by 0.63 mm) tooth, which has a machined flattened surface on the phonic wheel, to provide feedback of the angular position of any fan rotational imbalance.

Fan Air Seal

Behind the LP compressor disc is a LP Fan Air Seal which is bolted to the LP compressor shaft. The air seal contains a 5-fin labyrinth seal, the fins are subject to fan air pressure on one side and IP5 pressure on the other. The pressure drop across this seal is used to control bearing loads in the Front Bearing Housing. The seal also provides an inner aerodynamic line between the LP and IP compressors.



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LP COMPRESSOR

Spinner Assembly

Location

The spinner is located at the front of the engine and is attached to a support ring on the front of the LP compressor disc.

Purpose

The purpose of the spinner is to aerodynamically guide the air entering the engine into the LP compressor. It also allows any debris to be deflected to the outer section of the fan blades and down the bypass duct to protect the engine core.

Description

The spinner assembly consists of the spinner and spinner support ring.

Spinner

The spinner is a one piece, filament wound composite component that is conical in shape, painted black and coated with polyurethane to prevent erosion and minimise ice formation. A white line painted on the spinner from the tip rearwards provides an indication of engine rotation in low lighting conditions whilst the engine is running.

The tell tale rotation line starts at the tip just below the rubber anti-icing feature and corresponds with the # 1 fan blade position, it then spirals down the cone in an counter-clockwise direction twice and also ends at the # 1 fan blade position.

At the # 1 fan blade position on the base of the spinner the sequence of numbers 1, 2, & 3 are painted next to the corresponding fan blades and in addition an arrow points to the # 1 fan blade position, for ease of identification for the maintenance crews.

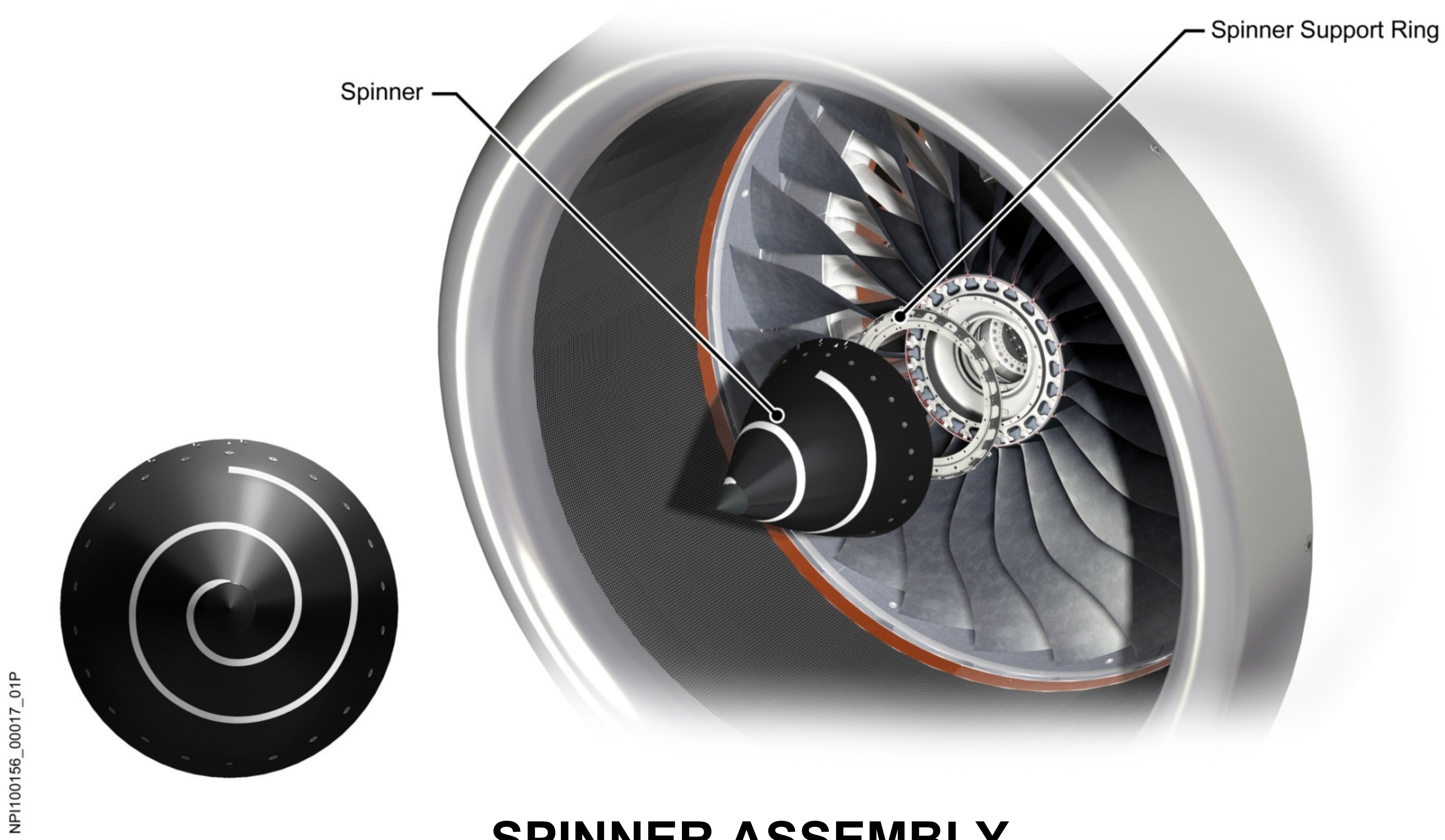
A rubber tip at the end of the spinner vibrates during engine running to act as an anti-icing feature and break up any ice that may form on the spinner. The outer surface extends rearwards to touch the annulus fillers and provide an aerodynamic cover for the support ring.

The spinner is attached to the nose cone support ring with 22 MORTORQ screws.

Spinner Support Ring

The spinner support ring is attached to the front of the LP compressor disc with 20 bolts. The support ring provides attachment provisions for dynamic balance weights.

Offset alignment pins fitted to the rear face of the support ring ensure the ring is only installed onto the disc in one position. Extraction inserts provide assistance in removal of the spinner support ring during maintenance.



SPINNER ASSEMBLY

LP Compressor Blade and Annulus Filler Assemblies

Introduction

The Trent XWB has 22 hollow wide chord, titanium, swept LP Compressor Blades designed to maximize efficiency and minimise noise.

Location

The LP Compressor Blades are located at the front of the engine attached to the LP Compressor Disc.

Purpose

The purpose of the LP Compressor Blades is to produce the majority of the thrust and to deliver air in a smooth flow for delivery into the IP compressor section.

Description

Each LP Compressor blade is manufactured by diffusion bonding two titanium plates to a central titanium membrane, which is then super-plastically formed to give the correct aerofoil shape. The leading edge of the blade is of elliptical shape to increase the efficiency of the blade.

Each blade locates into the LP Compressor disc by a curved dovetail root and retained axially by a shear key installed to the base of the blade. The shear key is retained to the fan blade by a flexible strap.

The blade root is laser peened and lubricated with a dry film lubricant to reduce surface stress between the blade roots and disc dovetail during operation. A slider assembly inserted between the dovetail slot of the disc and the base of the blades locates the blades radially and ensures the shear key is secured in position into the disc slot.

Etched on the bottom of each blade is the blades own specific information that includes:

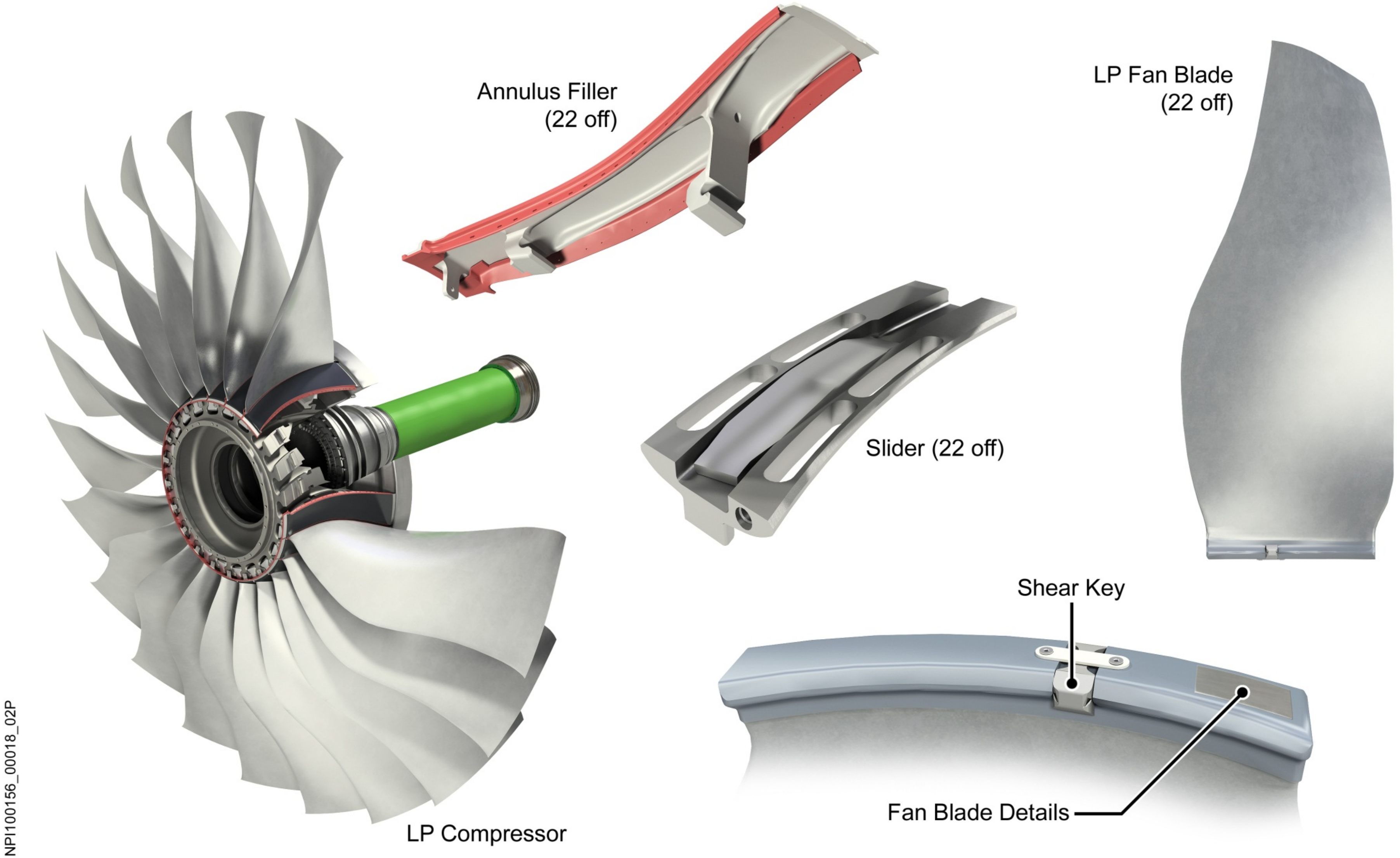
- The Assembly Part Number.
- The Serial No.
- The Tangential, Axial and Radial Moment Weights of the blade.

Annulus Fillers

22 aluminium annulus fillers provide a smooth aerodynamic profile between each fan blade into the compressors.

Each annulus filler has two hooks profiled in the casting these axially locate into position on the LPC disc, also a rear profile on the filler locates under the forward edge of the fan seal. To secure the forward edge of the filler, a single dowel pin provides location to the spinner support ring.

Rubber flap type seals on each side of the annulus filler minimises leakage of air from between the fan blade and annulus fillers.



FAN BLADE AND ANNULUS FILLER ASSEMBLIES

IP Compressor Module

Location

The IP compressor is located between the LP Compressor and the Intermediate Case.

Purpose

The purpose of the IP Compressor is to compress the air from the inner annulus of the LP Compressor, increase its pressure and deliver it at the correct conditions to the HP Compressor and to the EBAS.

Description

The IP compressor module is an eight-stage axial compressor assembly consisting of four main sections:

- Front Bearing Housing (FBH)
- The IP Compressor Stage 1-3 case.
- The IP Compressor Stage 4-8 case.
- The IP Compressor rotor.

Front Bearing Housing (FBH)

The FBH contains the LP location, IP compressor roller bearing the LP and IP systems speed probes. Around the outer annulus is hollow Engine Section Stators (ESS), which during certain conditions, are heated with hot air from the eighth stage of the IP compressor.

The ESS are welded together and fixed to the fan outlet guide vanes (OGV) to form the FBH / OGV joint.

The FBH / OGV joint hold the LP compressor case to the core engine. Electrical cables, from the shaft speed probes; pass internally through the ESS vanes. Other vanes contain tubes to supply oil to and from the bearings. Behind the ESS vanes is a single stage of Variable Inlet Guide Vanes (VIGVs).

IP Compressor Stage One to Four Case

The stage one to four case is split into two semi-circular half cases to aid production and repair. Two stages of Variable Stator Vanes (VSVs) are installed in the half casings that are connected to the VSV / VIGV mechanism. The two half casings are lined with abradable linings between the variable stators.

IP Compressor Stage Five to Eight Case

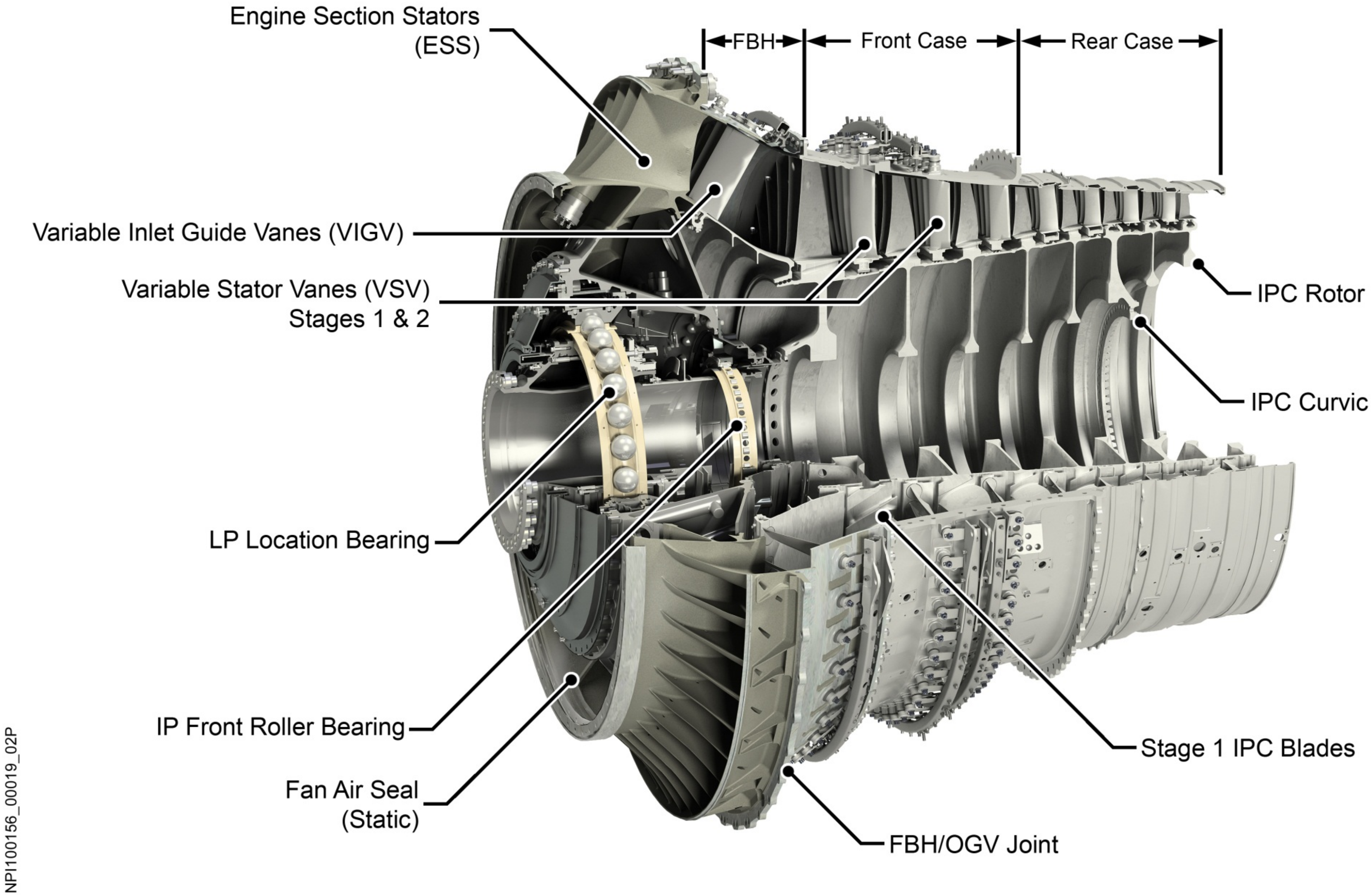
The IP Compressor Stage five to eight case is split into two semi-circular half cases, the forward flanged is bolted to the stage one to three case. The stage five to eight vanes are located in T slots, the stage eight vanes are known as the IP compressor outlet guide vane (OGVs). The two half cases are lined with abradable lining between the stator vanes.

IP Compressor Rotor

Eight discs are welded together to form a drum with the blades being mounted to the disc by axial and circumferential dovetail slots. All the blades run against abradable linings to help maintain tip clearances.

A stubshaft mounted to the front of the stage one rotor has a phonic wheel, with teeth machined into it, for measurement of the IP rotor speed (N2) and an inner race for the IPC front roller bearing.

To enable the IP Compressor to be rotated a drive arm at the rear of the stage six disc, is attached by a curvic coupling to the IP stub shaft within the compressor intermediate case module. Helical Spines within the IP stub shaft connects the IP Compressor to the IP Turbine driveshaft.



IP COMPRESSOR

Compressor Intermediate Case Module

Location

The compressor intermediate case is located between the IP compressor case and the HP system.

Purpose

The purpose of the compressor intermediate case is to provide:

- Support for the IP and HP compressor cases.
- A structure through which the thrust of the engine is transmitted to the aircraft.
- Location of the Internal and Intermediate Gearbox.
- ‘A’ frame location points.
- Locations of the IP handing bleed valve ports.

Description

The compressor intermediate case is a fabricated major structural part of the engine that forms the aerodynamic duct between IP & HP compressors. The module also provides, housing for the internal gearbox and a mounting position for the intermediate gearbox.

Intermediate Case

The front part of the intermediate case is installed over the rear part of the IP compressor and is bolted to a flange mid-way along the IP compressor case.

The rear part of the intermediate case is installed around the front part of HP compressor case and is bolted to a flange on the combustor outer case.

The outer casing incorporates a triangular box that features integral lugs. Thrust is transmitted from the engine to the rear mount and then to the aircraft via thrust links that are attached to the integral lugs of the intermediate case.

The rear flange of the triangular box features a V-groove that locates the thrust reverser inner surface in position when the thrust reverser assembly is closed around the engine.

Four LP compressor case supports (‘A’ frames), are attached to the intermediate case by four lugs to provide torsional stability between the intermediate case and the LP Compressor case.

The inner structure of the intermediate case incorporates the internal gearbox housing, which is supported by eight hollow aerofoil shaped struts. Some of the struts provide locations for tubes that take oil to and from the internal gearbox housing. Others supply air to the internal areas for cooling and sealing. There is also instrumentation within two other struts.

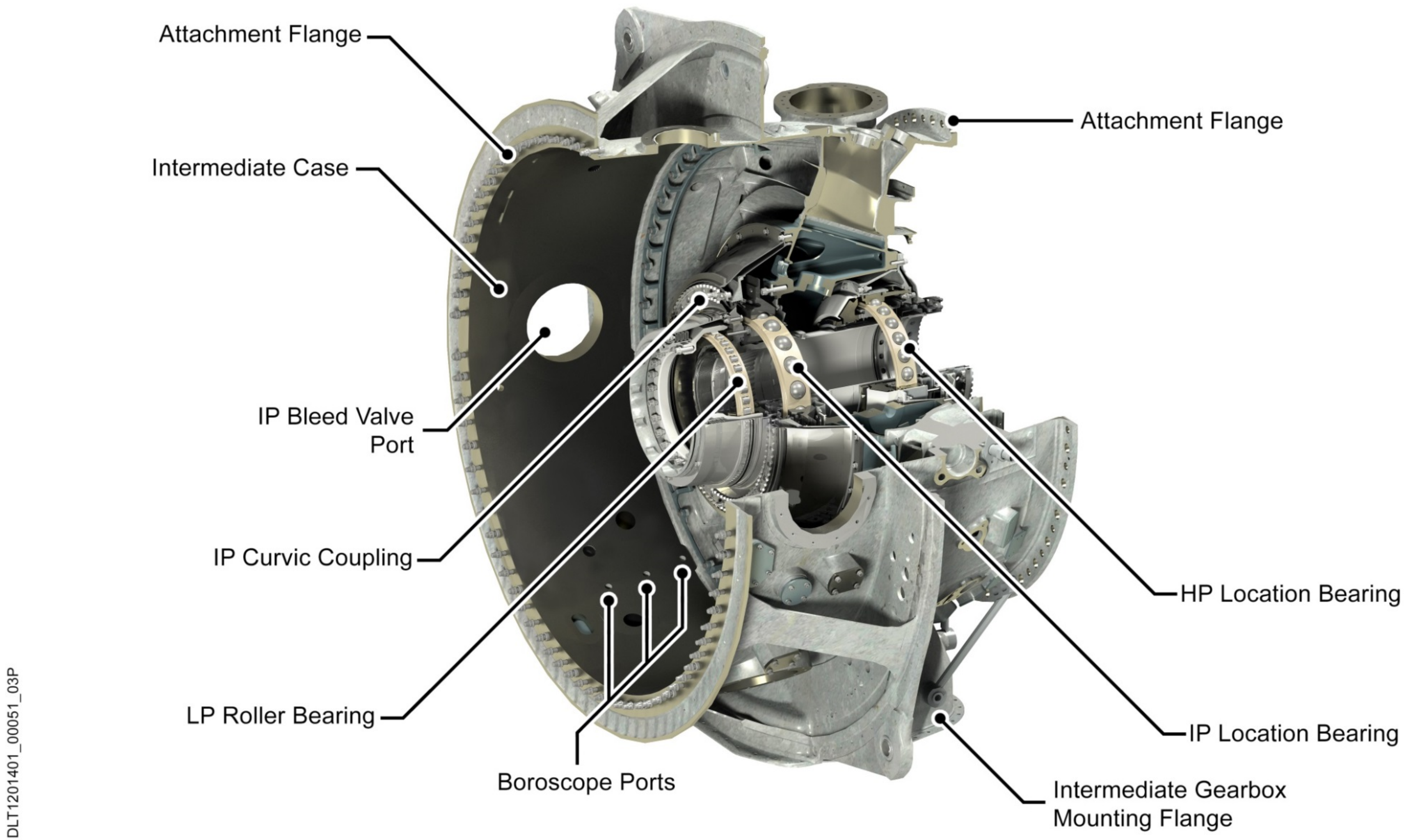
Internal Gearbox

The internal gearbox housing contains the LP roller bearing and IP and HP location bearings for the three rotating assemblies.

There are also gears from the HP rotating systems that drive the Intermediate Gearbox. This allows the HP rotating system to drive the external gearbox and to be turned for maintenance.

Intermediate Gearbox

Provision is made at the 6 o’clock position for the location of the intermediate gearbox. The intermediate gearbox transfers drive from the HP system to the external gearbox through a radial driveshaft in the 6 o’clock strut in the compressor intermediate case.



INTERMEDIATE CASE

High Pressure (HP) System Module

Location

The HP system module is located between the Intermediate Case and the Intermediate Pressure (IP) Turbine.

Purpose

The primary purpose of the HP system is to efficiently deliver compressed air at the required conditions for combustion. It also converts the combustion gases into a rotating force using a single stage HP turbine. The HP system secondary purposes are to drive the external gearbox and deliver compressed air to the EBAS.

Description

The HP system comprises three sub-assemblies:

- HP compressor.
- Combustor section.
- HP turbine.

HP Compressor

The HP compressor rotor comprises of three sub-assemblies:

- Stage 1-3 drum.
- Stage 4 disc.
- Stage 5, 6 drum and cone assembly.

Stages 1-3 are of blisk construction, and welded together to form a drum assembly.

Stage 4 is a conventional bladed disc to which the stage 1-3 drum and stage 5, 6 drum and cone assembly are bolted.

Stages 5 and 6 are conventionally bladed discs that are welded together into a drum and cone assembly. The cone incorporates a mini disc to which the HPT drive shaft connects.

Definition: A blisk is a single component consisting of a rotor disk and blades manufactured from a single piece of material.

Combustion Section

The combustion chamber is an annular design with tiled inner and outer walls. The tiles are coated with a Thermal Barrier Coating (TBC) to protect the base metal from heat in the combustor and cooled by HPC stage 6 air.

At the rear of the combustor are High Pressure Nozzle Guide Vanes (HPNGV) assemblies. Each vane is coated with TBC and cooled by HPC stage 6 air. The HPNGVs allow the gas flow from the combustor to enter the HP turbine at the correct velocity, angle of attack and direction.

Located on the single skin combustion outer case, are the 20 Fuel Spray Nozzles (FSN) and the HPC handling bleed valve ports.

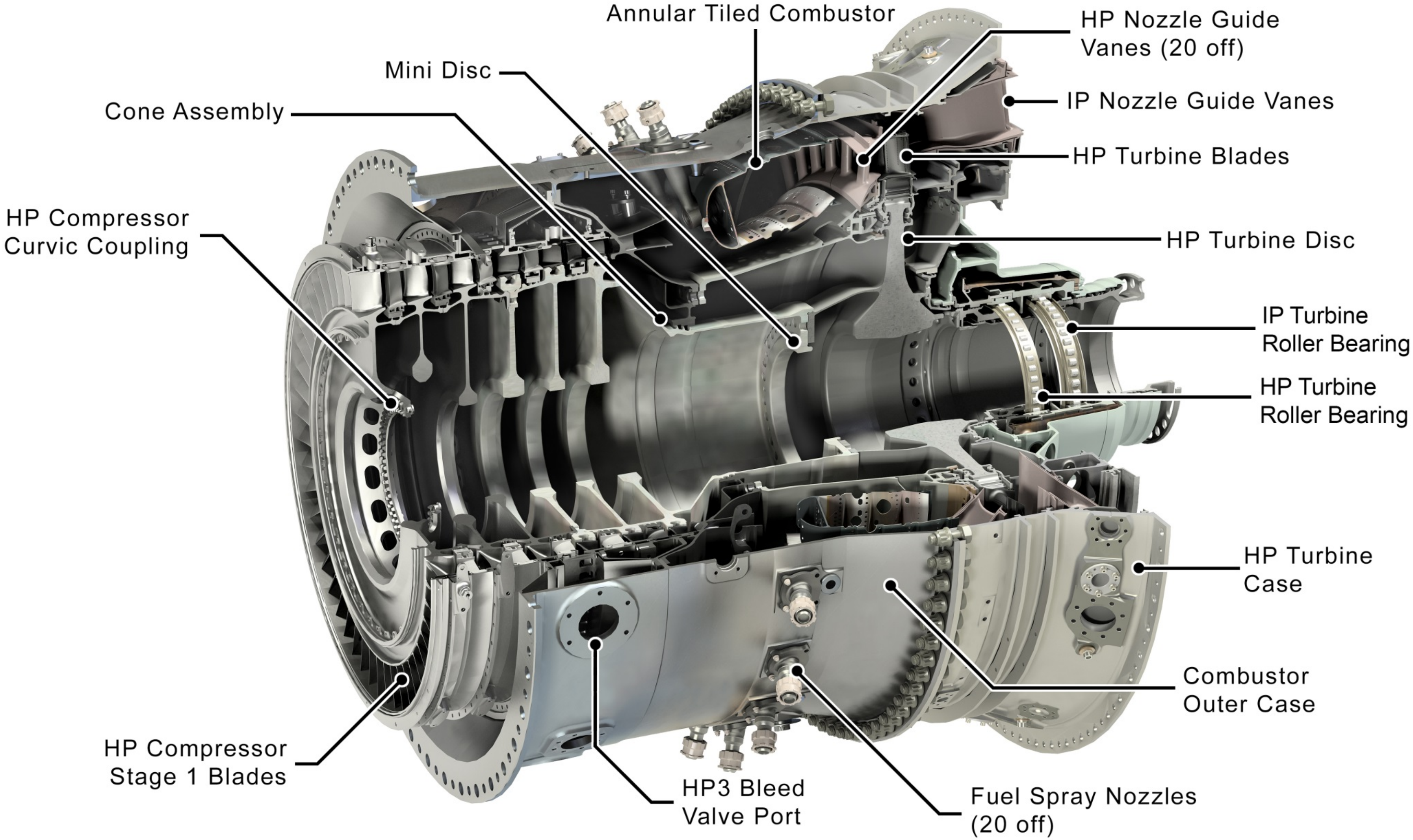
HP Turbine

The HP turbine blades are installed into the HP Turbine disc by fir-tree type root fixtures. The blades are internally cooled by HPC stage 6 air and have TBC applied to the blade surfaces. The disc is bolted at the front to a mini disc and at the rear to a stubshaft that extends rearwards supported by the HP roller bearing.

The HP turbine case is bolted to the outer combustion case and IP turbine case. A cooling manifold allows fan air to cool the outer casing and reduce clearance between the HPT blade tip and the seal segments located in the HPT case.

At the rear of the HP Turbine is the HP / IP bearing structure, this also contains the IP stage 1 NGVs which are cooled by HP3 air. Two thermocouples (TCAF 1 & 2) are located in two of the IPT stage 1 NGVs, monitoring the air temperature in the cavity between the HPT and IPT stage 1 disc.

This module also incorporates the IPT stage 1 NGVs.



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HP SYSTEM

IP Turbine (IPT) Module

Location

The IPT module is located between the HP system and LP turbine.

Purpose

The purpose of the IPT is to extract energy from the gas flow exiting the HPT, to drive the IP compressor.

Description

The IP turbine module is a two stage rotating assembly consisting of the following sub-assemblies:

IP Turbine Case - The single piece IP turbine case provides location for:

- IPT stage 2 NGVs.
- LPT stage 1 NGVs.
- IPT stage 1 and 2 seal segments.
- IP Turbine Case-Cooling manifold and modulating air valve.

IP Turbine Stage 2 NGVs - The IPT stage 2 NGVs are manufactured in pairs and are located between the stage 1 and 2 IPT blades. HPC stage three air is used to internally cool the first stage NVGs and only the inner platform of the second stage NVGs

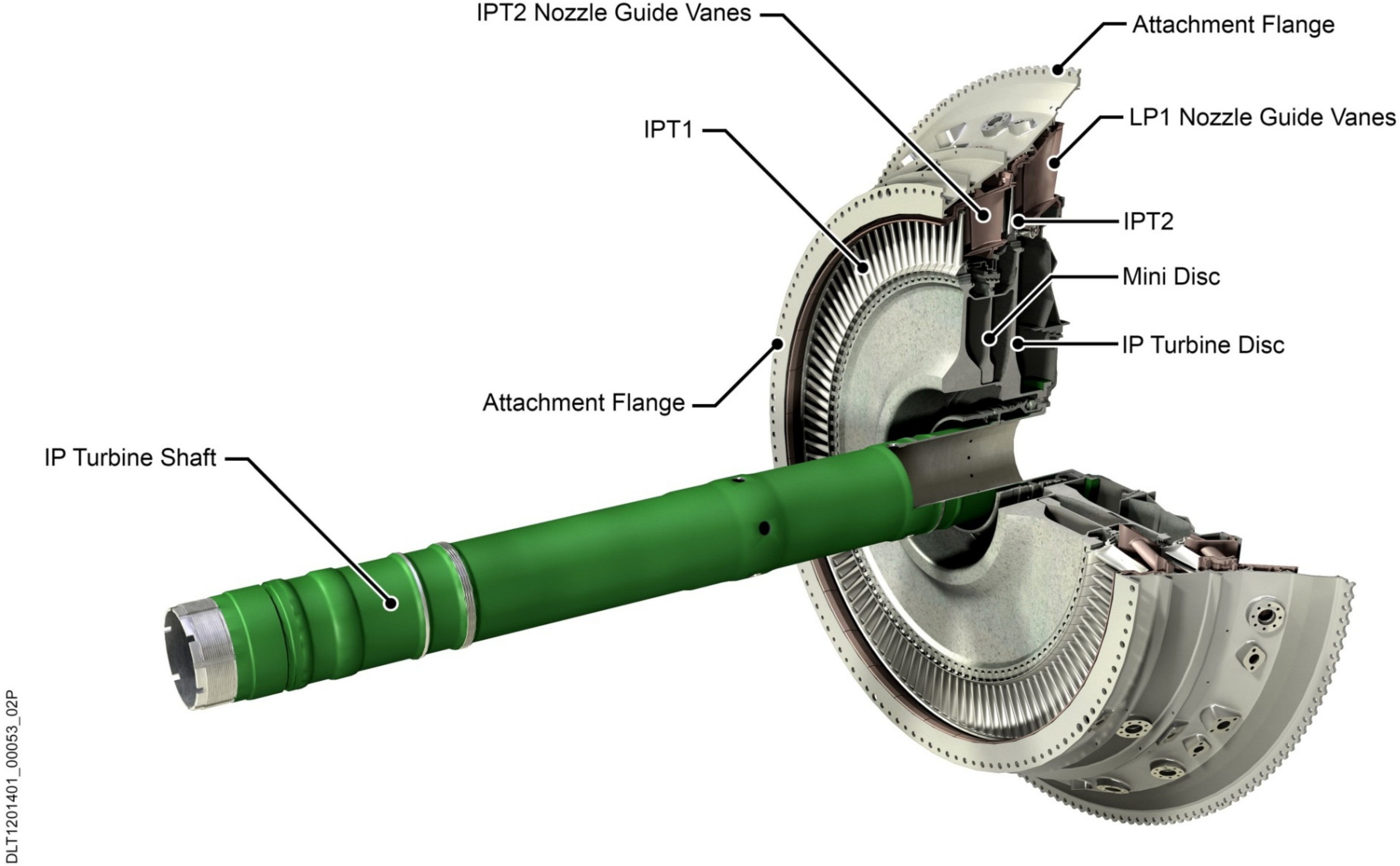
IP Turbine Blades - There are two stages of IPT blades, which are secured to the discs by fir-tree roots. The blades have shrouded tips with fin seals that run in seal segments. HPC stage three air is used to internally cool the first stage blades and only the fir tree roots of the second stage blades.

IP Turbine Discs - There are three discs within the IPT module:

- **IPT Stage 1 Disc** – The IPT stage 1 disc has two rearward facing arms. The outer arm bolts to a mini disc and the inner arm locates to the IPT stage 2 disc.
- **IPT Stage 2 Disc** – The IPT stage 2 disc has two forward facing arms and a rear drive arm. The front outer arm bolts to the mini disc and the front inner arm locates to the IPT stage 1 disc. The rearwards facing arm is bolted to the IPT stubshaft transferring energy from the turbine blades to drive the IP Compressor.
- **IPT Mini-Disc** – The primary purpose of the IPT mini-disc is to provide a fin seal at the base of the IPT stage 2 NGVs to prevent gas leakage. The secondary purpose is to provide strength to the IPT 1 and 2 interface.

LP Turbine Stage 1 NGVs - Located at the rear of the IPT module are the LP1 NGVs. They are cooled internally by IPC stage 8 air. Located in 12 of the NGVs, are thermocouples that measure the temperature of the gas flowing through the turbine section. This temperature is indicated on the flight deck as Engine Gas Temperature (EGT). Near to top dead centre, a separate thermocouple (TCAR) passes through one NGV to monitor the air temperature at the rear of the IPT 2 disc.

IP Turbine Stubshaft - The IP turbine stubshaft provides the inner race of the IP rear roller bearing to provide radial support for the IP System.


IPT2 Nozzle Guide Vanes
Attachment Flange
LP1 Nozzle Guide Vanes
IPT1
IPT2
Mini Disc
IP Turbine Disc
Attachment Flange
IP Turbine Shaft
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IP TURBINE

LP Turbine (LPT) Module

Location

The LPT module is located at the rear of the engine after the IPT module.

Purpose

The purpose of the LPT is to extract energy from the gas flow exiting the IPT to drive the LP compressor (Fan).

Description

The LP turbine module is a six stage rotating assembly consisting of the following sub-assemblies:

LP Turbine Case and Stage 2 to 6 NGVs

The LPT case provides location for the LPT Stage 2 to 6 NGVs and seal segments.

The outer casing is cooled with fan air which is supplied to LPTCC manifold via a two position solenoid controlled air valve.

LP Turbine Blades

The LPT blades are attached to the discs by fir-tree roots and have circumferential seal-fins attached to the outer tip shrouds. The seal fins run in seal segments to control gas leakage. Stages one and two are solid in construction and stages 3 to 6 are partially hollow for weight reduction.

LP Turbine Discs

Six individual LPT discs are bolted together and feature circumferential seal fins to control the cooling air and hot gas leakage in the LPT. Fir-tree roots allow the turbine blades to be located to the discs. The stage 5 disc incorporates a drive arm that attaches to the LPT shaft via a bolted curvic coupling.

LP Turbine Shaft

The LPT shaft is connected to the front of the LPT stage five disc by a bolted curvic coupling. The shaft goes through the centre of the IP shaft to connect with the LPC shaft by helical splines and a threaded nut.

LP Turbine Stubshaft

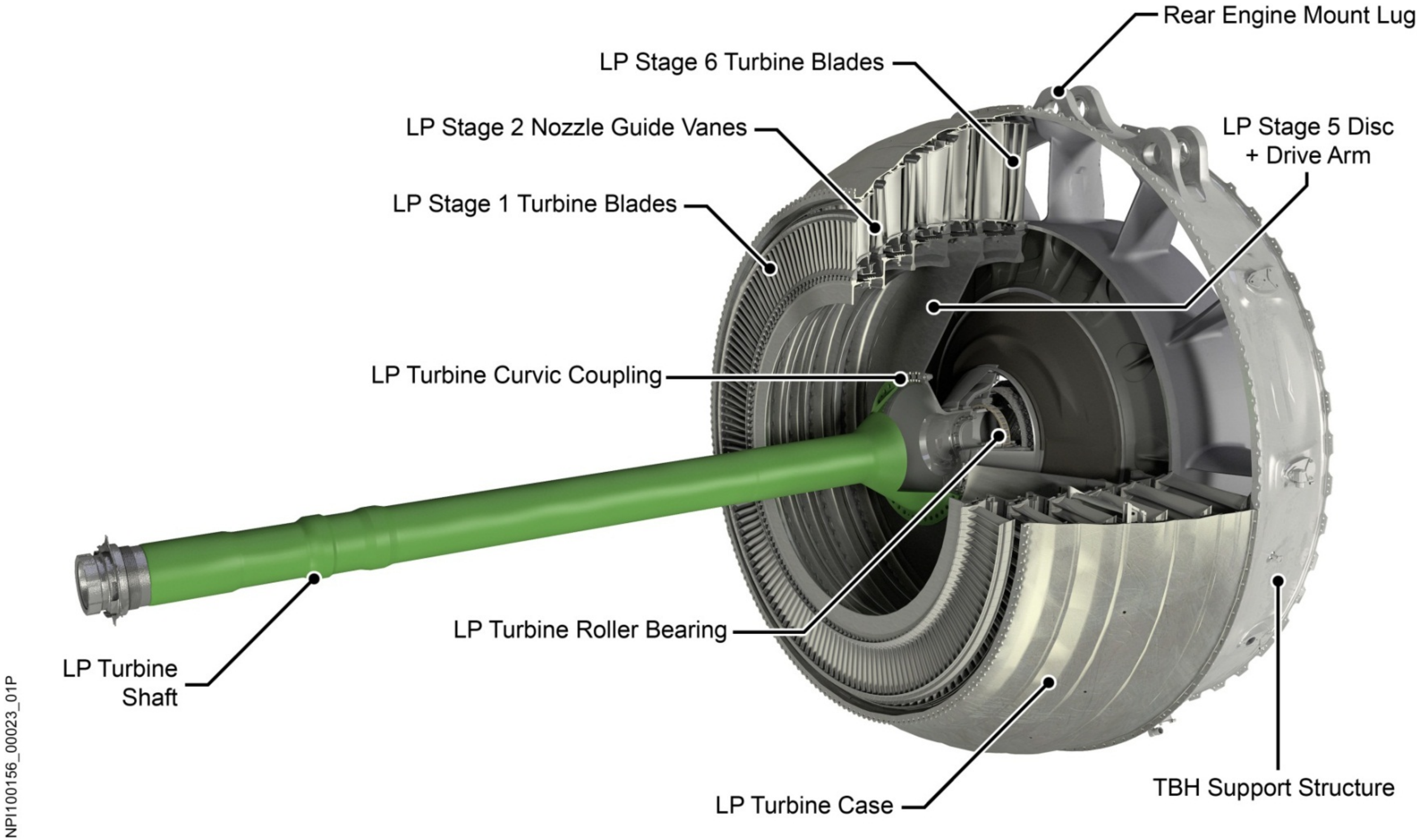
The LP turbine stubshaft is bolted to the rear of stage 5 disc and provides radial support for the LPT roller bearing in the TBH. A phonic wheel at the rear of the stubshaft provides indication of LPT speed to the Engine Electronic Controller (EEC).

Tail Bearing Housing (TBH)

The TBH is bolted to the rear of the LPT case and provides a location feature for the hot gas exhaust system. The outer case has integral lugs for the attachment of the rear engine mount.

Located in the centre of the TBH is the LPT rear roller bearing to provide radial support for the LP System. Four speed probes interact with the LPT stubshaft phonic wheel to provide an LPT speed signal to the EEC. The bearing housing is held concentrically to the outer case by twelve radial hollow struts that straighten the gas flow and contain IP8 air and oil servicing tubes to the bearings.

A conical air seal maintains the IP8 air pressure around the bearing housing and a heat shield inside the air seal protects the bearing housing from excessive temperatures.


Rear Engine Mount Lug
LP Stage 6 Turbine Blades
LP Stage 2 Nozzle Guide Vanes
LP Stage 5 Disc + Drive Arm
LP Stage 1 Turbine Blades
LP Turbine Curvic Coupling
LP Turbine Roller Bearing
LP Turbine Shaft
TBH Support Structure
LP Turbine Case
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LP TURBINE

Engine Transmission System

Introduction

The HP system is used to provide a mechanical drive for external accessories; it also provides a means to rotate the engine HP system during the start cycle and for maintenance procedures. The following components make up the engine transmission system:

- Internal gearbox
- Intermediate gearbox
- Radial drive shaft
- External gearbox (EGB)

Internal gearbox

Location

The internal gearbox is part of the intermediate module.

Purpose

To provide a rotating drive from the HPC shaft to the intermediate gearbox.

Intermediate Gearbox

Location

The Intermediate Gearbox is attached to the intermediate case module at the 6 o'clock position.

Purpose

The purpose of the intermediate gearbox is to transfer the drive from the HP system to the external gearbox. This is achieved by taking a drive from the HP system through two matched helical gears (Internal Gearbox), which then provides drive to the external gearbox by the Radial driveshaft.

Radial drive shaft

Location

The Radial drive shaft is located in the lower interservices (Splitter) fairing between the intermediate gearbox and the external gearbox.

Purpose

To provide a mechanical drive between the gearboxes

External Gearbox

Location

The external gearbox is located and mounted to the bottom of the LP compressor case.

Purpose

To provide a mounting point and mechanical power for the engine driven accessories and aircraft systems.

Description

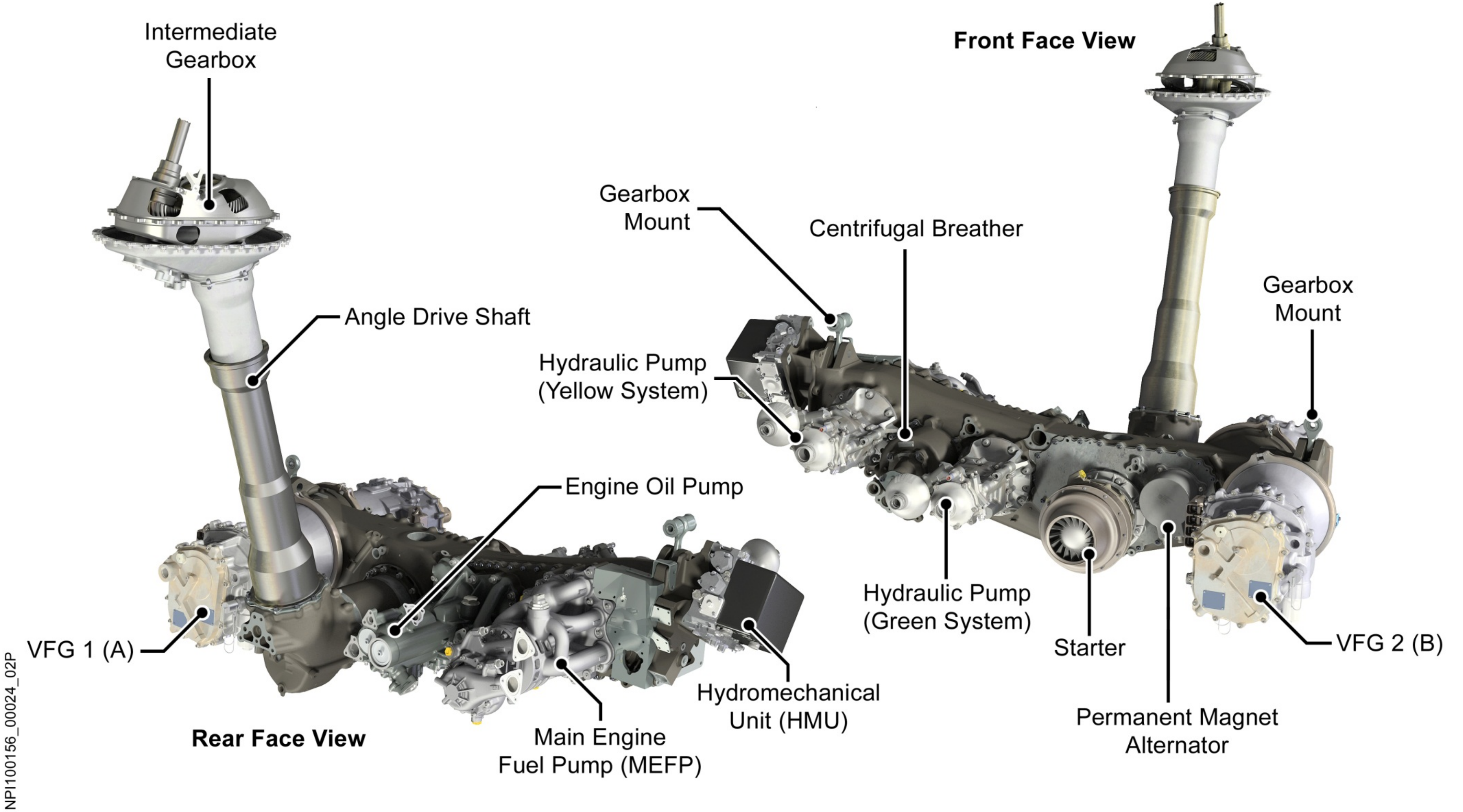
Power from the HP system is transmitted via the intermediate gearbox and an Radial drive shaft to the external gearbox. The power is used by the accessory gear train inside the gearbox to drive a total of eight accessory units plus the centrifugal breather. The eight components are described later in this section.

During a ground engine start, power is transmitted to the HP system from the air turbine starter motor, through the EGB and to the Intermediate Gearbox via a drive shaft.

The centrifugal breather housing provides a means of hand turning the HP rotor system for maintenance procedures.

All the accessory interfaces are protected by a drains system that removes any leaking fluids from the gearbox area to prevent build-up of any hazardous fluid and the risk of fire.

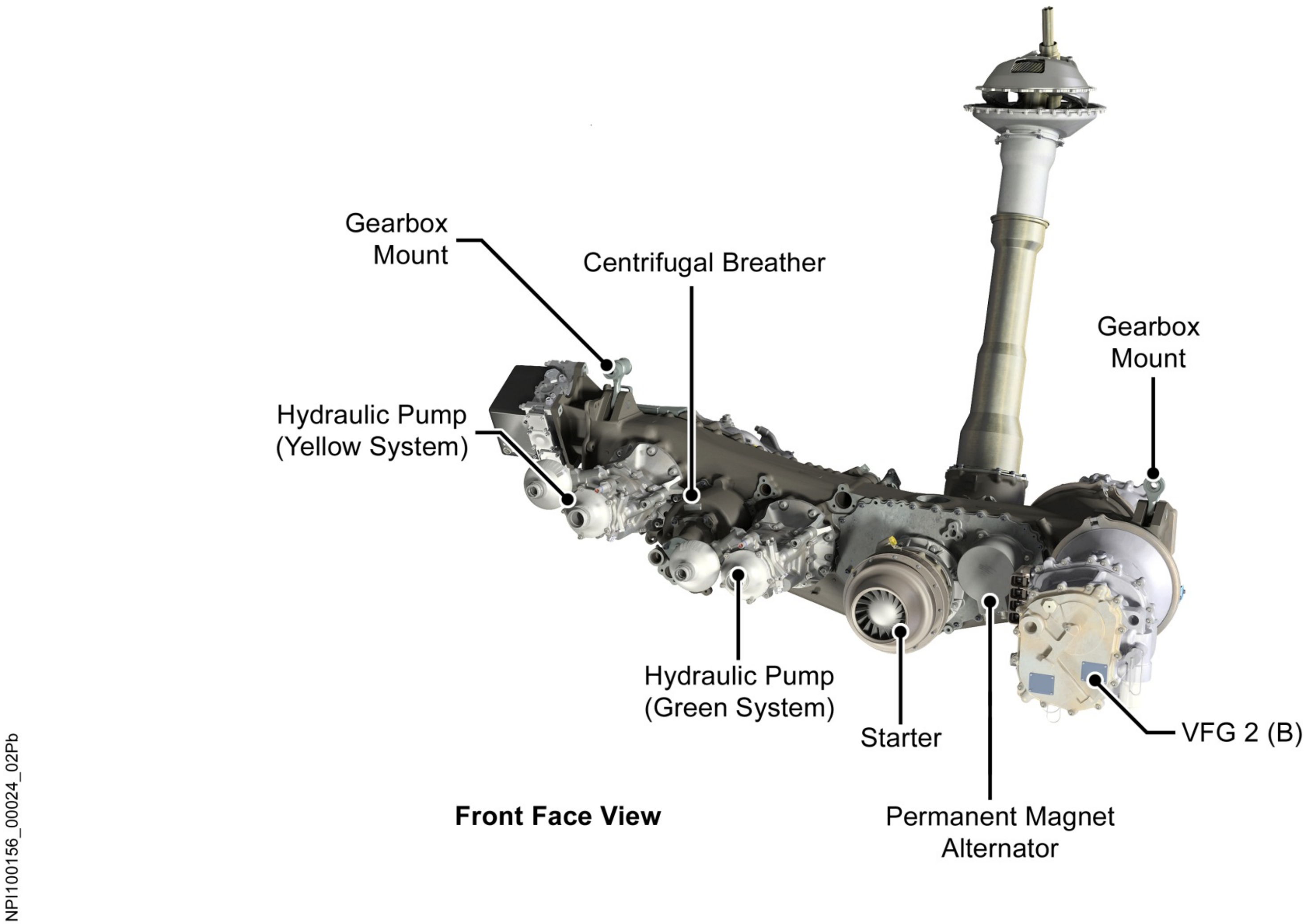
The Hydro-Mechanical Unit (HMU) is mounted to an adaptor block on the right side of the external gearbox and is connected to the LP/HP fuel pump assembly.



ENGINE TRANSMISSION SYSTEM

Components Mounted on the Front Face

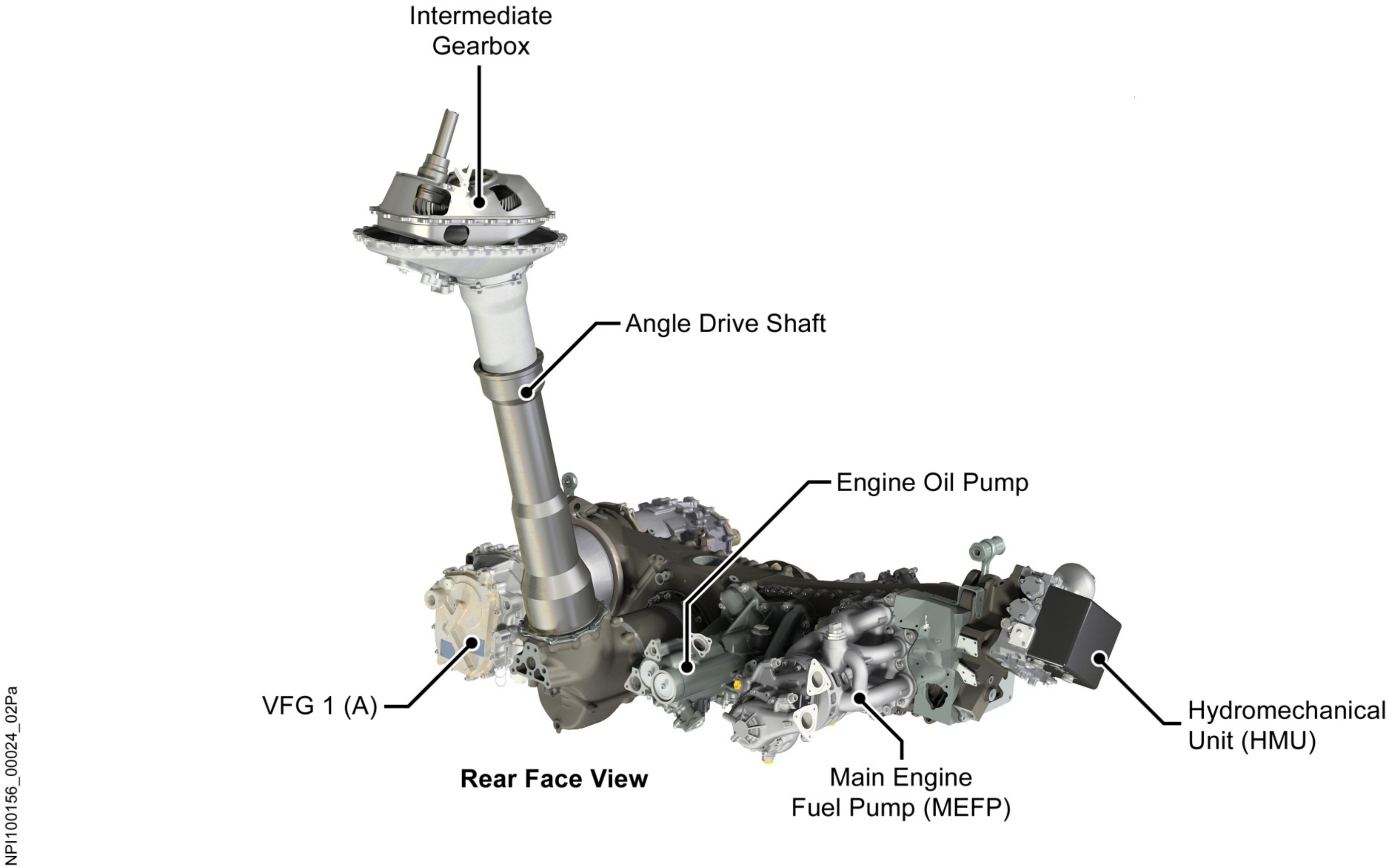
- Permanent Magnet Alternator (PMA).
- Variable Frequency Generator (VFG-Front).
- Air Turbine Starter Motor.
- One yellow system hydraulic pump.
- One green system hydraulic pump.
- Centrifugal Breather.



COMPONENTS MOUNTED ON THE FRONT FACE

Components Mounted on the Rear Face

- Variable Frequency Generator (VFG-Rear).
- Oil Pump Assembly.
- Main Engine Fuel Pump Assembly (MEFP).



COMPONENTS MOUNTED ON THE REAR FACE

LP Compressor Case (Non modular)

Location

The LP compressor case is at the front of the engine and covers the LP compressor.

Purpose

The purpose of the LP compressor case is to

- To optimize the pressure and velocity of the fan airflow.
- Withstand the extreme force of a fan blade release.
- Reduce noise from the fan when the engine is running.
- Provide a mounting position for engine components.

Description

The LP compressor casing assembly consists of four main sub-assemblies and is a non-modular assembly:

- The LP compressor front case (LPC FC).
- The LP compressor rear case (LPC RC).
- The LP compressor mounting ring and outlet guide vane (LPC OGV) assembly.
- LP compressor supports.

LP Compressor Front Case (LPC FC)

The LPC FC is at the front of the LP compressor case and is a titanium-designed ring with circumferential stiffening ribs that provide reinforcement to the fan track area in the unlikely event of an LP compressor blade release.

The inner surface of the containment case features on-wing replaceable front and rear acoustic panels to absorb noise generated by the fan and a fan track / ice impact panel for which the fan is able to cut its own seal to prevent fan tip air losses.

The LPC FC is bolted to the LP compressor mounting ring.

LP Compressor Rear Case (LPC RC) and the LPC Case Supports

The composite LPC RC is bolted to the rear of the outer mount ring. A titanium V-groove ring is bolted to the rear of the LPC Rear Case to provide a structural reaction fixture for the thrust reverser assembly.

The inner surface of the rear fancase has apertures for Surface Air Oil Heat Exchangers (SAOHE) for the engine and Surface Air Cooled Oil Coolers (SACOC) for the VFG oil systems.

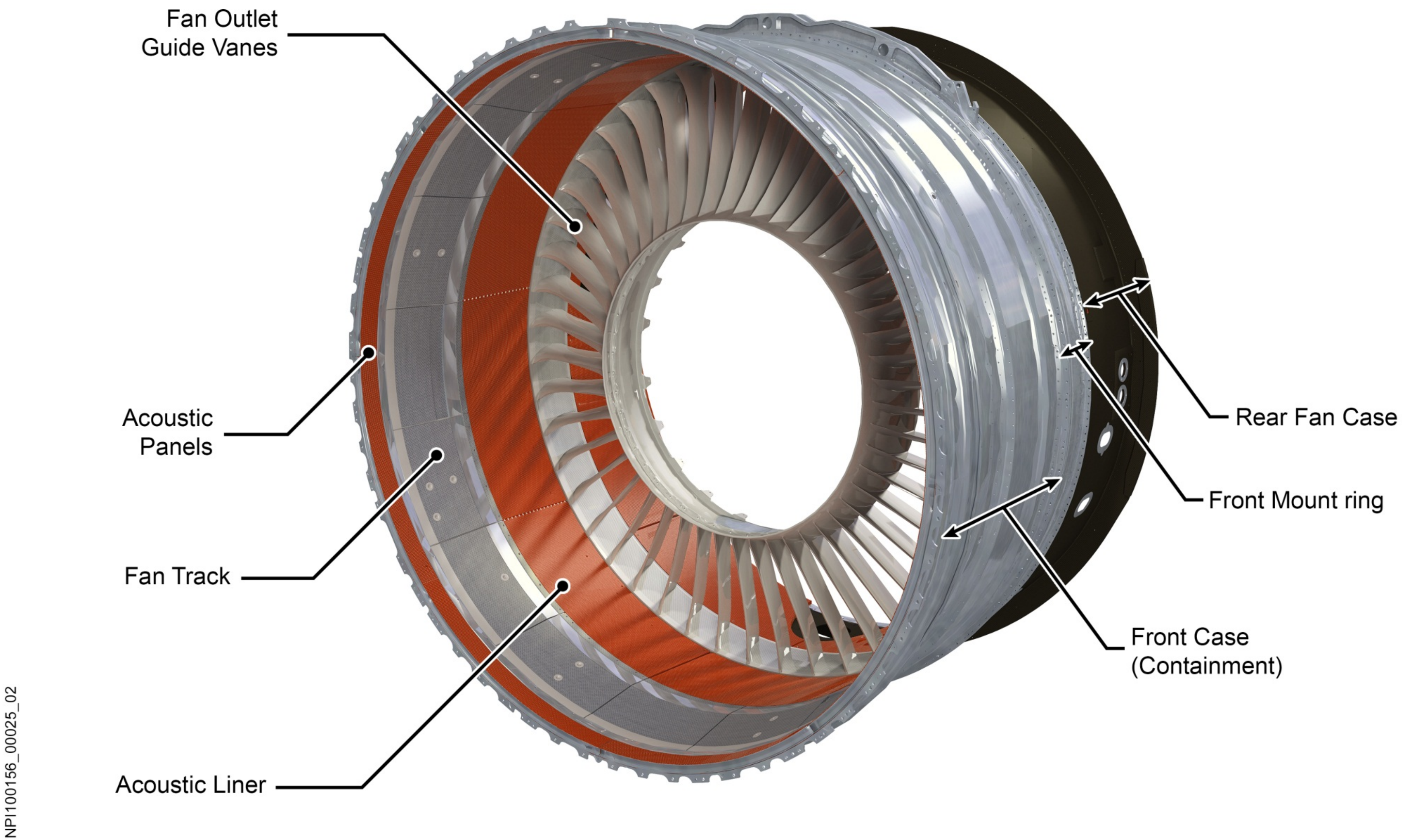
Acoustic panels are also secured to the inside surface of rear fan case to absorb noise from the LP compressor.

LPC case supports (A frames) provide torsional stiffness between the LP compressor rear case and the engine core. They have an aerodynamic fairing over them and provide a route for electrical harnesses and other services between the LP compressor case and the core.

LPC Mounting Ring and Outlet Guide Vane (LPC OGV) Assembly

48 titanium Outlet Guide Vanes are welded to an inner mount ring that forms the core to fancase interface. The assembly bolts to the outer mount ring to provide the radial support for the LP compressor case and aerodynamic control to the airflow entering the bypass duct.

The outer surface of the outer mount ring also provides mounting points for the external gearbox and the forward engine mount that is located at the top of the outer mount ring.



LP COMPRESSOR CASE (NON MODULAR)

Engine Components – Looking Left Side

The following components are located on or are visible on the left side of the engine:

Fire Zone 1 (Fan Case)

- Engine Electronic Controller (EEC).
- Data Entry Plug (DEP).
- Engine Monitoring Unit (EMU).
- Two Igniter Boxes.
- Rear VFG.
- Forward VFG.
- Forward Engine Mount.
- Inlet Cowl Anti-Ice Valve.
- Start Air Valve.
- Two Engine Surface Air Cooled Oil Coolers (SACOC) for VFG Oil.
- Two Oil Thermal Bypass Valves.

Zone 2 (Under the Gas Generator Fairings)

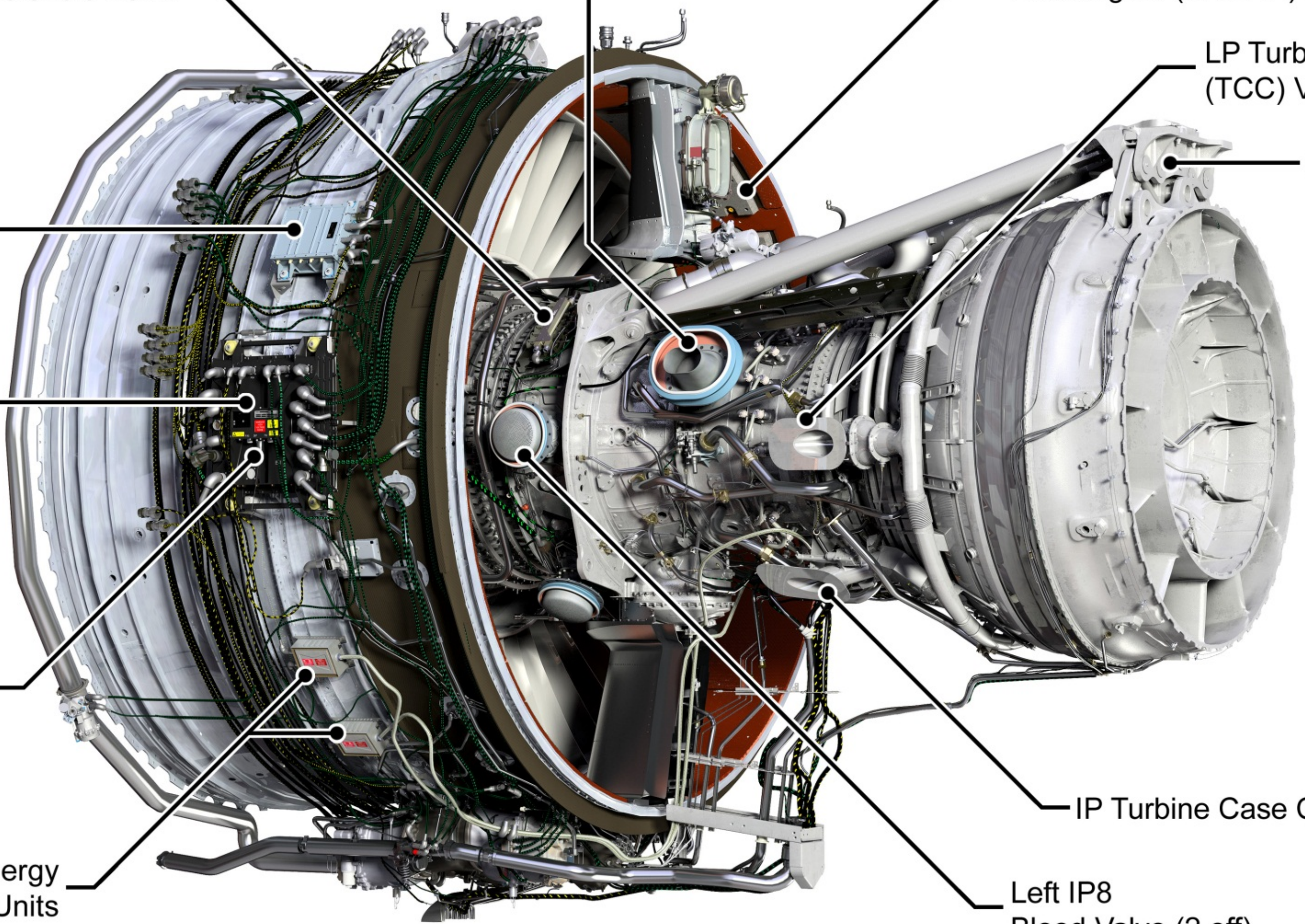
- Two IP8 Engine Handling Bleed Valves.
- Left Solenoid Bank.
- Left VIGV/VSV Actuator.
- Zone two Fire and Overheat Detectors.

Zone 3 (The Core)

- One HP3 Engine Handling Bleed Valve.
- Left Igniter Plug.
- Ten Fuel Spray Nozzles & Fuel Manifold.
- IP Turbine Case Cooling Valve (IP TCCV).
- LP Turbine Case Cooling Valve (LP TCCV).
- Six Turbine Gas Temperature (TGT) Thermocouples.
- Rear Engine Mount and Thrust Links.

Underside

- External Gearbox.
- Drains Mast.



ENGINE COMPONENTS – LOOKING LEFT SIDE

Engine Components – Looking Right Side

The following components are located or are visible on the right side of the engine:

Fire Zone 1 (The Fan Case)

- Oil tank, including the Oil Quantity Transmitter.
- Oil Debris Monitoring System (ODMS) - Includes the Scavenge Filter Housing incorporating the Oil Debris Sensor and the Scavenge Filter ΔP Transducer.
- Engine Fuel Oil Heat Exchanger (FOHE) - includes the LP Fuel Filter, the LP Fuel Filter Δp Transducer, Low Oil Pressure Switch and Oil Differential Pressure Transducer.
- Engine Oil Bypass Valves (OBV).
- Engine Fuel Temperature Sensor.
- Engine Oil Temperature Sensors.
- Forward Engine Mount.
- Fuel Flow Transmitter.

Zone 2 (Under the Gas Generator Fairings)

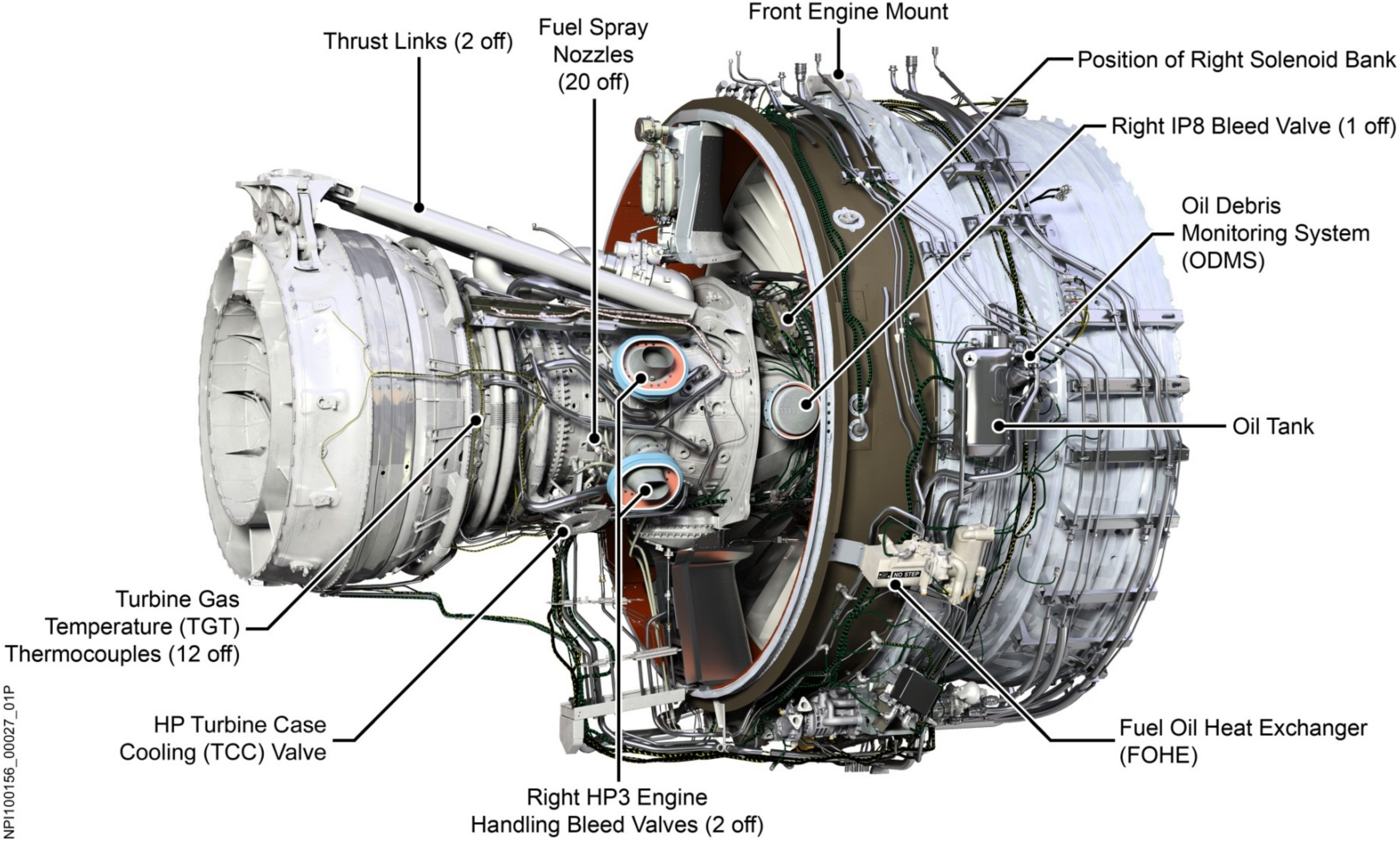
- One IP8 Engine Handling Bleed Valve.
- Right Solenoid Bank.
- Right VIGV/VSV actuator.
- Zone two Fire and Overheat Detectors.
- Engine Section Stators (ESS) Anti-Ice Valve and Manifold.

Zone 3 (The Core)

- Two HP3 Engine Handling Bleed Valves.
- Right Igniter Plug.
- Ten Fuel Spray Nozzles & Fuel Manifold.
- Six Turbine Gas Temperature (TGT) Thermocouples.
- Rear Engine Mount and Thrust Links.
- HP Turbine Case Cooling Valve (HP TCCV).
- Vibration Transducer.

Underside

- External Gearbox.
- Drains Mast.



ENGINE COMPONENTS – LOOKING RIGHT SIDE

Borescope Access

Location

The Trent XWB engine has a total of 28 borescope access ports along the length of the right side of the engine mainly between the 3 and 6 o'clock positions.

Purpose

The purpose of the borescope ports is to allow internal inspection of the internal components of the gas path for inspection using borescope equipment.

LP Compressor

There are no borescope access ports on the LP Compressor as it can be visually inspected from the front of the engine.

IP Compressor

There are 8 borescope access ports for the IP Compressor:

Stage 1 leading edge – Through the front of the IP Compressor.

IPC1/2 – Trailing edge stage 1 / leading edge stage 2.

IPC2/3 – Trailing edge stage 2 / leading edge stage 3.

IPC3/4 – Trailing edge stage 3 / leading edge stage 4.

IPC4/5 – Trailing edge stage 4 / leading edge stage 5.

IPC5/6 – Trailing edge stage 5 / leading edge stage 6.

IPC6/7 – Trailing edge stage 6 / leading edge stage 7.

IPC7/8 – Trailing edge stage 7 / leading edge stage 8.

IPC 8 – Trailing edge stage 8.

HP Compressor

There are 5 borescope access ports for the HP Compressor:

HPC1 – Leading edge stage 1 only.

HPC1/2 - Trailing edge stage 1 / leading edge of stage 2.

HPC2/3 – Trailing edge stage 2 / leading edge stage 3.

HPC3/4 – Trailing edge stage 3 / leading edge stage 4.

HPC4/5 – Trailing edge stage 4 / leading edge stage 5.

Combustion Chamber / HP Nozzle Guide Vanes (HP NGV)

There are 6 borescope access ports positioned around the circumference of the combustion outer case to allow inspection of the combustion chamber and the HP NGVs.

HP Turbine.

A single borescope port (HPT) accesses the space between a pair of HP NGVs to allow the HP turbine blade leading edge to be inspected.

IP Turbine

There are two IP turbine access ports:

HPT/IPT1 – Trailing edge HPT / leading edge IPT 1.

IPT1/2 – Trailing edge IPT 1 / leading edge IPT 2.

LP Turbine

There are 6 borescope access ports for the LP Turbine:

IPT2/LPT1 – Trailing edge IPT2 / leading edge LPT1.

LPT1/2 – Trailing edge LPT1 / leading edge LPT2.

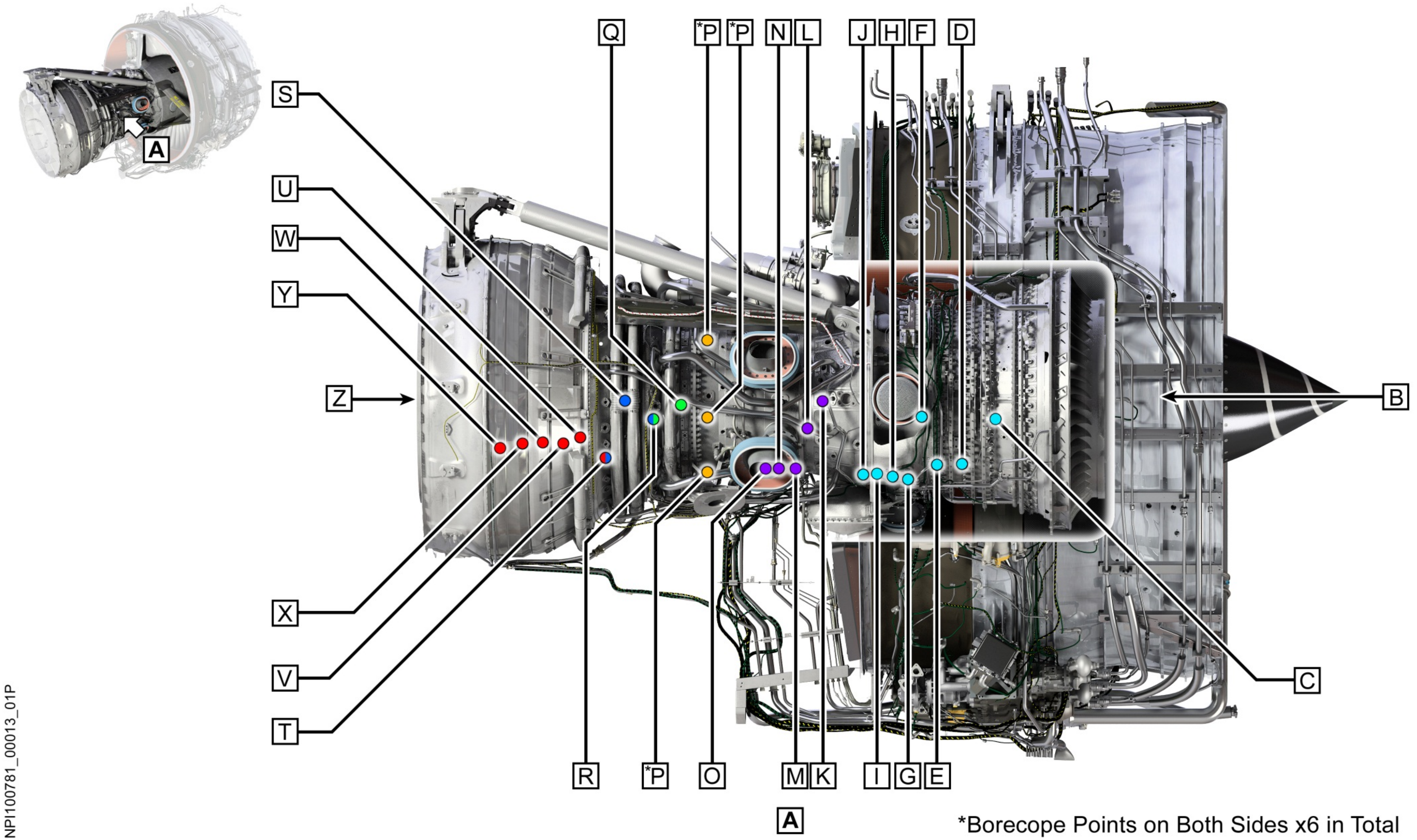
LPT2/3 – Trailing edge LPT2 / leading edge LPT3.

LPT3/4 – Trailing edge LPT3 / leading edge LPT4.

LPT4/5 – Trailing edge LPT4 / leading edge LPT5.

LPT5/6 – Trailing edge LPT5 / leading edge LPT6.

The trailing edge of LPT stage 6 is accessed through the exhaust.



BORESCOPE ACCESS POSITIONS

Boroscope Port Plugs

There are several different types of Boroscope plugs on the Trent XWB engine depending on their location.

There are shorter double sealing plugs on the IPC area these are made up of an inner and an outer seal to ensure no gas escapes from the inner or outer engine sections.

On the HPC the plugs are also of a double plug type but due to depth of the core engine the plug is longer than those on the IPC.

Due to the length of the HPC plugs there are several designs to take in account the thermal expansion and stress loads that the long plug stems are subjected to.

The first of these design features is an articulated stem arm that is free to float in the external plug head and is allowed to float around on the internal plug area.

Another design is a sealed piston that is half way down the plug shaft, when the plug is inserted into the correct location the piston fits into a mid-span sleeve to form a seal to secure the plug in position, this allows for thermal axial movement.

For the combustion area there are several access points these are made up of two ignitor ports and x 6 ¼ inch square drive plugs.

The plugs are identified by being yellow in colour and the ports have their name etched on the engine external case adjacent to the plug port.

Blanking Plugs

Where appropriate, the impact extractor & adaptor may be used to assist with removal of the plugs.

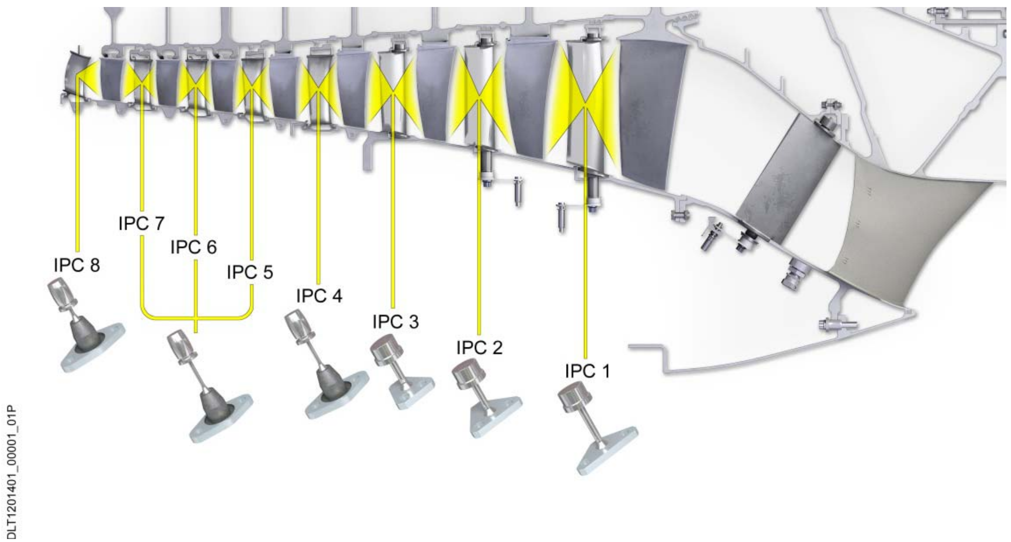
Although incorrect fitting of plugs is designed to be impossible, it is an advantage to identify them to their respective ports.

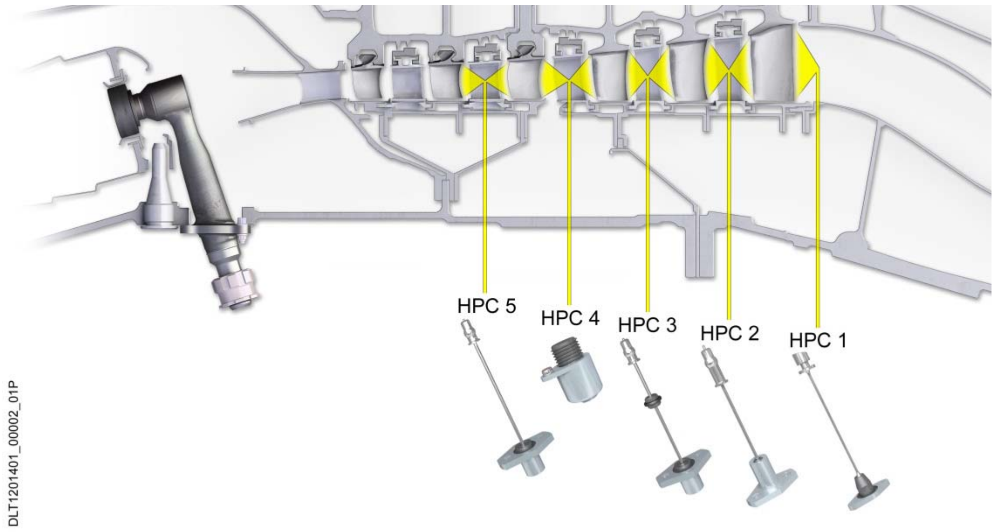
Note

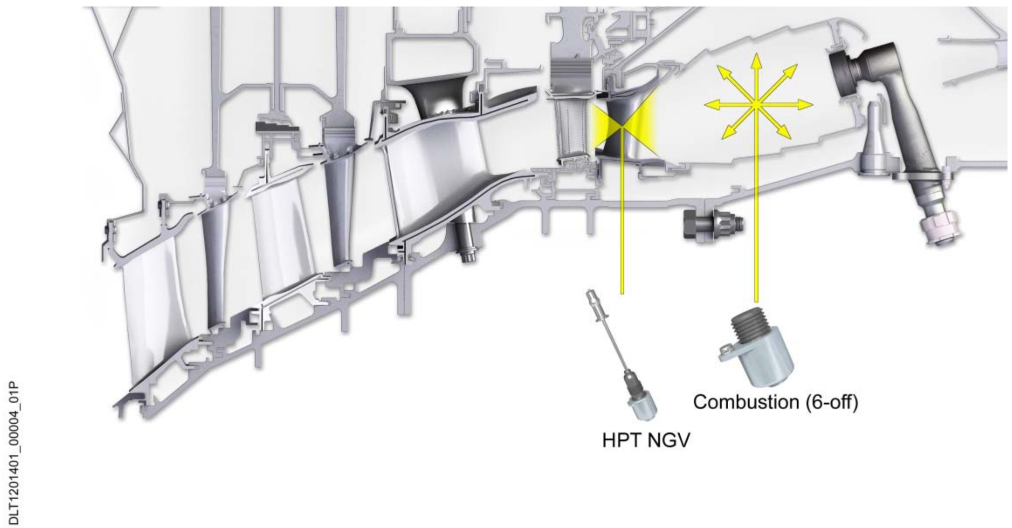
On completion of the inspection, the plugs should be refitted in accordance with AMP instructions, paying particular attention to torque loading of securing bolts, and the application of the correct anti-seize compound applied to the surfaced as instructed by the relevant AMP.

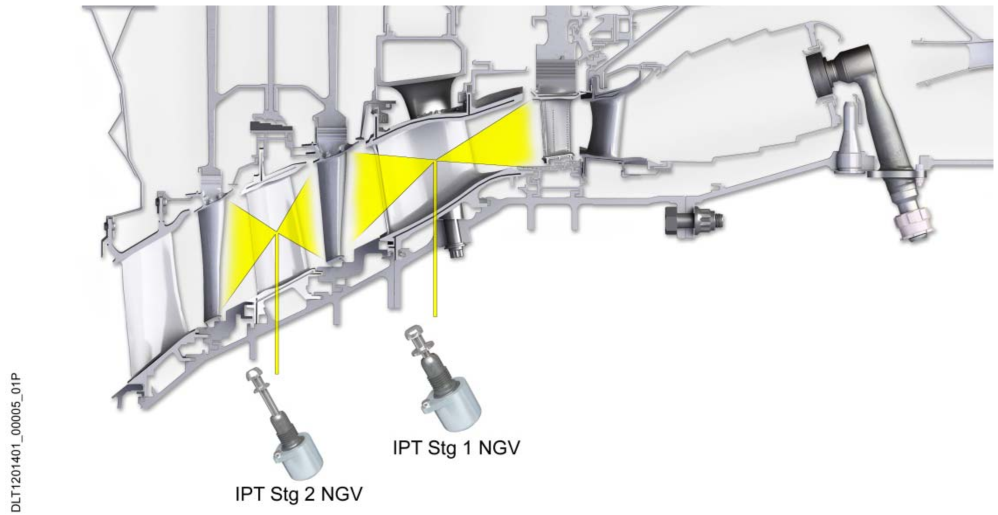
CAUTION

NEVER USE THE SECURING BOLTS TO PULL THE PLUGS INTO POSITION. THIS MAY RESULT IN DAMAGE TO THE ENGINE OR THE PLUG.









TYPES OF BOROSCOPE PORTS PLUGS

Maintenance Practices
The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialized tooling.

LP Compressor Blades Visual Examinations Job Set-Up

Aircraft maintenance configuration –
TRENTXWB-A-72-31-13-00A01-722C-D

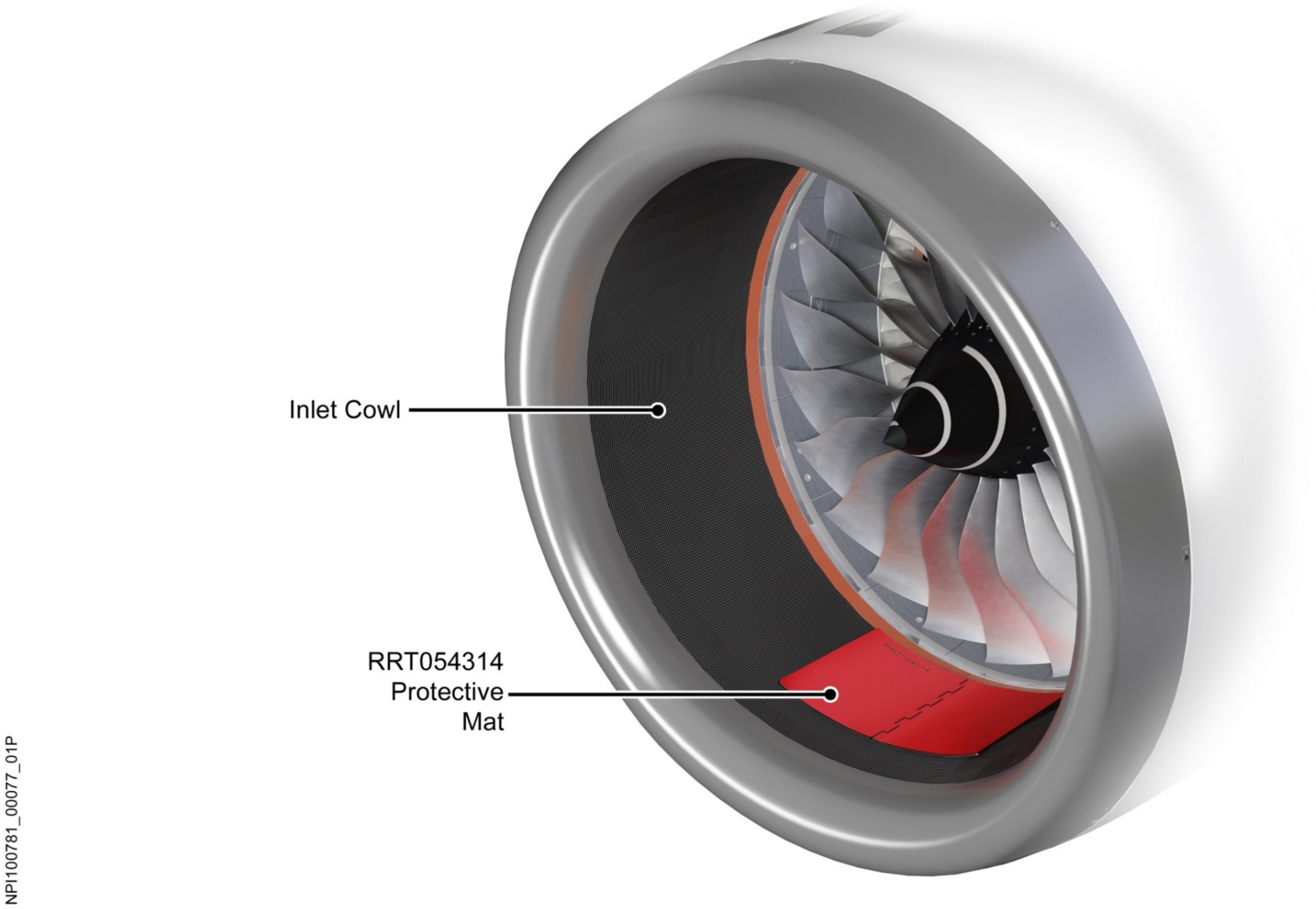
Safety precautions –
TRENTXWB-A-72-31-13-00A01-722C-D

Make sure that the engine 1, (2) shutdown occurred not less than 5 minutes before you do this procedure.

Open the applicable circuit breaker(s) –
TRENTXWB-A-72-31-13-00A01-722C-D

Procedure
1. Put the Mat (RRT054314) in the inlet cowl.
2. Install the Immobilizer (RRT061241),

Refer to TRENTXWB-A-72-31-13-00A01-722C-D



INLET COWL PROTECTIVE MAT

TURNING THE LOW PRESSURE (L.P.) SYSTEM (AMP 72-00-00)

WARNING

YOU MUST BE CAREFUL WHEN YOU DO WORK ON THE ENGINE PARTS AFTER THE ENGINE IS SHUT DOWN. THE ENGINE PARTS CAN STAY HOT FOR ALMOST 1 HOUR.

WARNING

YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE INJURY. IF YOU GET AN INJURY PUT IT INTO COLD WATER FOR 10 MINUTES AND GET MEDICAL AID

WARNING

MAKE SURE THE APPLICABLE COVERS ARE INSTALLED TO THE REAR OF THE ENGINE. THE MOVEMENT OF AIR THROUGH THE ENGINE CAN CAUSE THE L.P. COMPRESSOR TO TURN VERY QUICKLY AND CAUSE INJURY.

WARNING

YOU MUST USE APPLICABLE GLOVES ON YOUR HANDS WHEN YOU HOLD THE LP COMPRESSOR BLADES. THE LEADING EDGES OF THE BLADES CAN CAUSE AN INJURY

Turning the LP System

You must go into the air intake cowl to turn the L.P. system which can be turned by hand.

Procedure:

- Position a suitable access platform in a safe position and install the Exhaust Nozzle and Thrust Reverser Covers.
- Position a suitable access platform in a safe position at the Engine Air Intake Cowl. And install the inlet protective rug into position in the air intake cowl. Make sure red warning flag of the mat can be seen externally of the intake cowl.
- Enter the intake cowl. And turn the L.P. compressor with your hand.

When task is complete ensure all equipment tools and fixtures are removed.

TURNING THE LOW PRESSURE (L.P.) SYSTEM

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Fan Set Immobilization Job Set-Up

Aircraft maintenance configuration –
TRENTXWB-A-72-31-13-00A01-722C-D

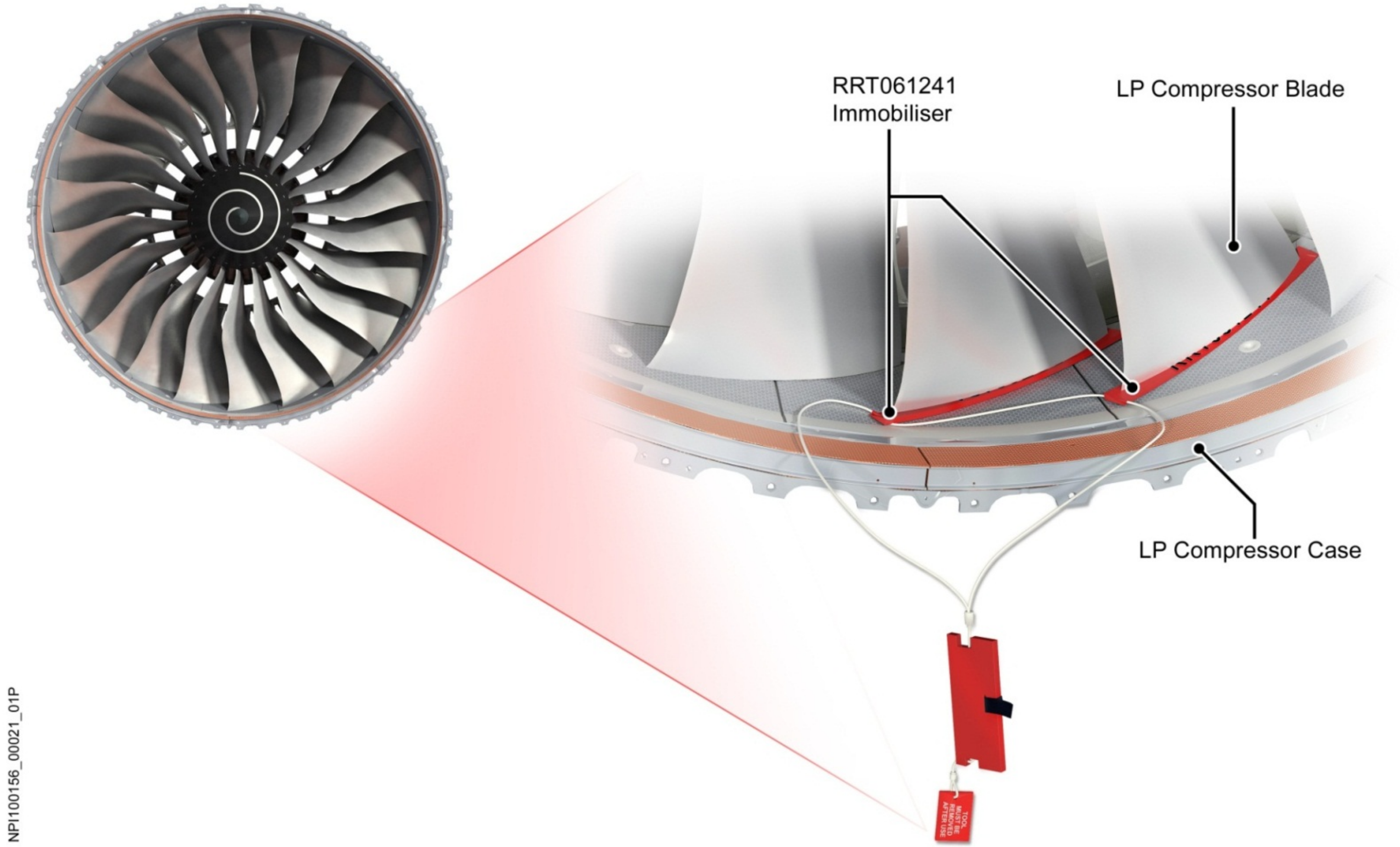
Safety precautions –
TRENTXWB-A-72-31-13-00A01-722C-D

Make sure that the engine 1; (2) shutdown occurred not less than 5 minutes before you do this procedure.

Open the applicable circuit breaker(s) –
TRENTXWB-A-72-31-13-00A01-722C-D

Get access
Put the applicable access platform into a safe position at the left side of the engine.
Put the applicable access platform into a safe position at the right side of the engine.

Procedure
Install the Immobilizer RRT061241;
Refer to TRENTXWB-A-72-31-13-00A01-722C-D.



FAN SET IMMOBILISATION

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

LPC Fan Blade Inspections

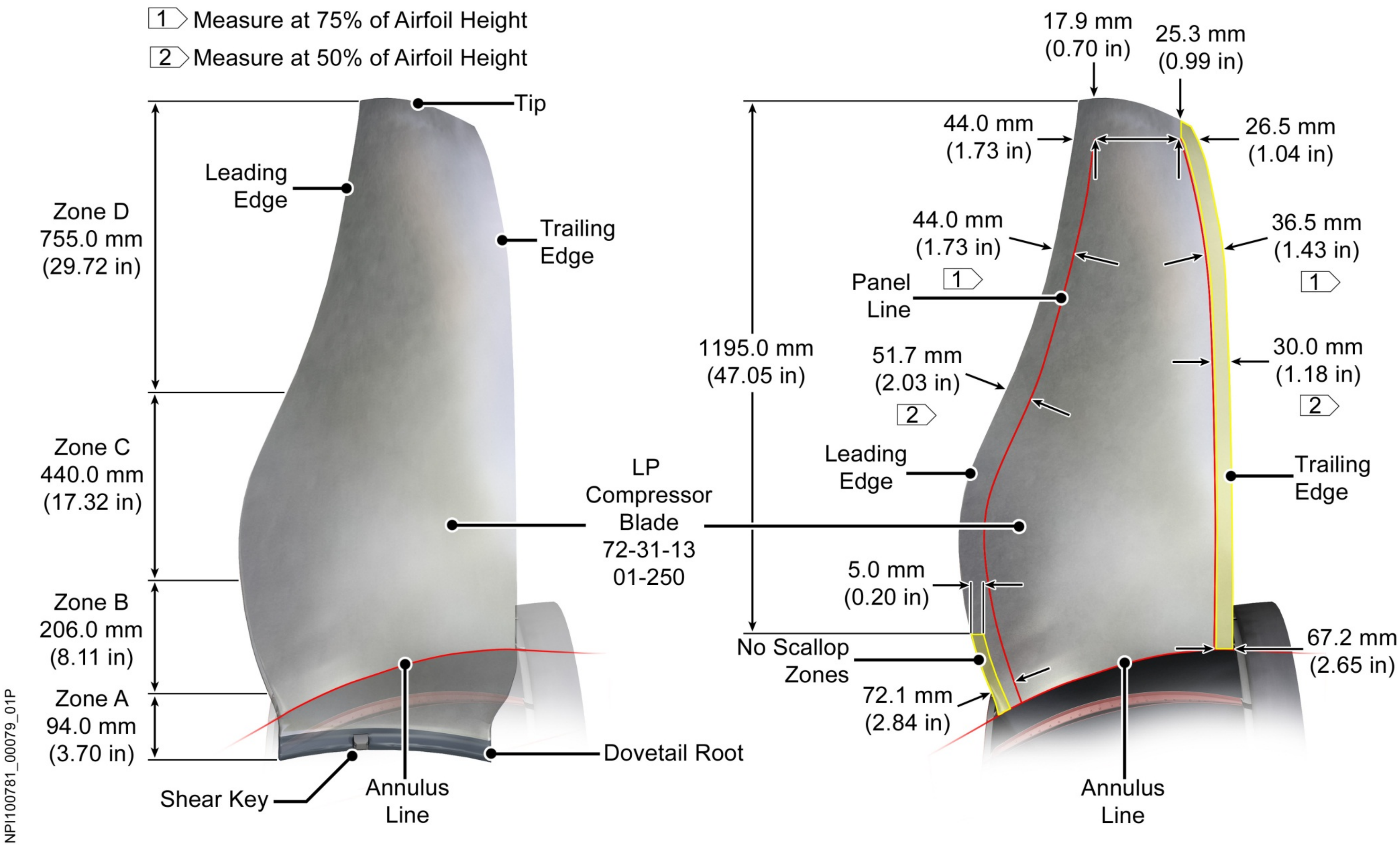
Examine each of the 22 LP Compressor blades as follows.

- Examine the aerofoil surfaces for cracks.
- Examine the aerofoil surfaces for rippling.
- Examine the tip and the aerofoil surfaces for heat discoloration (blueing).
- Examine the aerofoil surface for arc-burns.
- Examine the aerofoil surfaces for scratches.
- Examine the leading edge for nicks.
- Examine the trailing edge for nicks.
- Examine the leading edge and the trailing edge for bends.

– The bend does not extend into the panel area.

If the bend has a circumferential dimension ‘A’ that is more than 0.197 in. (5.00 mm) but not more than 0.346 in. (8.80 mm), do the steps that follow.

If the dimension of the bend is in the limits that follow, replace the LP Compressor blade.



LPC FAN BLADE INSPECTION

Maintenance Practices

LPC Fan Blade Inspections

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

If there are more than three bent LP Compressor reject the entire bent LP Compressor blade.

If there is more than one bend in one LP Compressor blade reject the LP Compressor blade.

If the bend is not smooth or has kinks, tears, cracks or nicks, reject the LP Compressor blade.

If there is untwist, reject the LP Compressor blade.

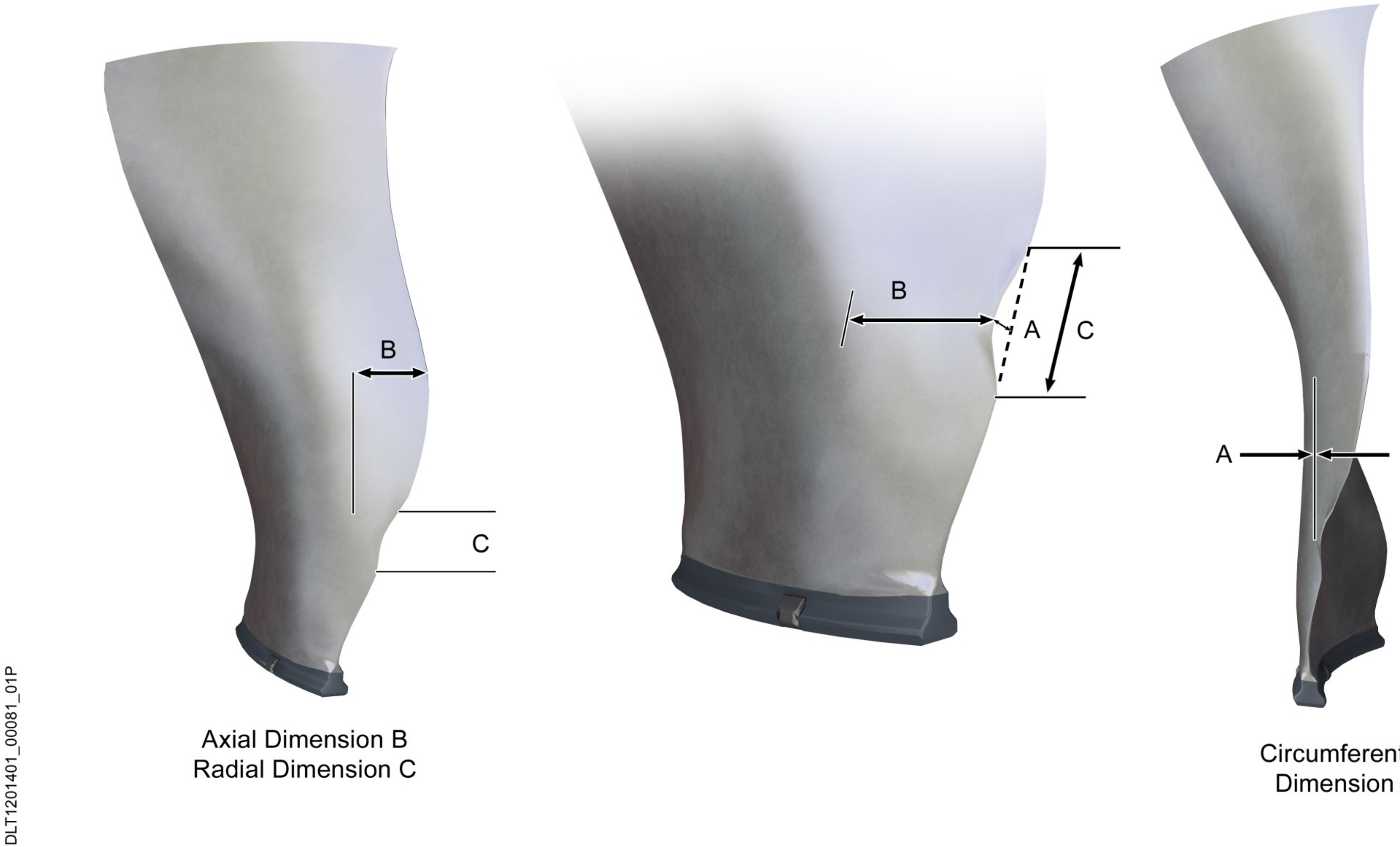
Replace the LP Compressor blade before not more than 125 flight hours or 25 flight cycles.
Use the first of the flight hours or flight cycle limits to occur:

– The axial dimension B is more than eight times the length of circumferential dimension A.
– The radial dimension C is more than 15 times the length of the circumferential dimension A.
– The bend does not extend into the panel area.
If the dimension of the bend is not in the limits, reject the LP Compressor blade.

If the bend has a circumferential dimension A that is more than 0.346 in. (8.80 mm), reject the LP Compressor blade.

Remove the Immobilizer RRT061241,
Refer to TRENTXWB-A-72-31-13-00A01-522C-D.
Remove the Mat RRT054314 from the inlet cowl.

Close-Up
Make sure that the work area is clean and clear of tools and other items.
Close the applicable circuit breaker(s) –
TO BE SUPPLIED BY AIRBUS INDUSTRIE.



Axial Dimension B
Radial Dimension C

Circumferential
Dimension A

LPC FAN BLADE INSPECTIONS

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Turning of IPC Job Set-Up

Aircraft maintenance configuration –

TRENTXWB-A-72-00-00-00A01-950C-A

Safety precautions –

TRENTXWB-A-72-00-00-00A01-950C-A

Make sure that the engine 1; (2) shutdown occurred not less than 5 minutes before you do this procedure.

Open the applicable circuit breaker(s) –

TRENTXWB-A-72-00-00-00A01-950C-A

Get access.

1. Put the applicable access platform into a safe position at the left side of the engine.
2. Put the applicable access platform into a safe position at the right side of the engine.

Procedure

1. Install the Immobiliser RRT061241;
2. Refer to TRENTXWB-A-72-31-13-00A01-722C-D.
3. Turn the IP system.
4. Install the Turning tool (HU43985-2).
5. Put the Turning tool (HU43985-2) through the LP compressor blades, the engine section stator and the VIGVs.

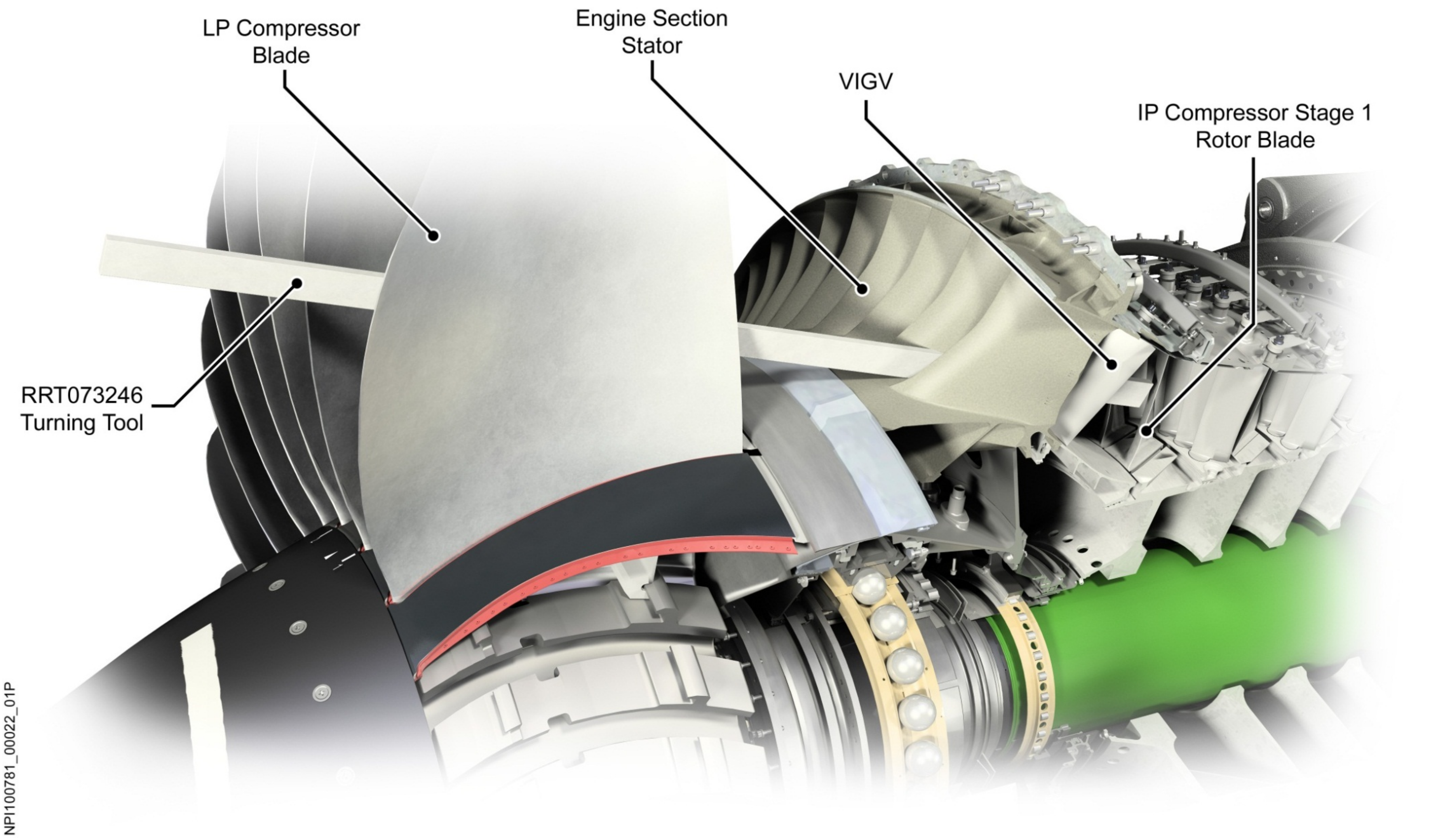
If necessary, turn the LP compressor blades to get access to the IP compressor stage 1 rotor blade,

Refer to TRENTXWB-A-72-00-00-00A01-950C-A.

6. Use the Turning tool (HU43985-2) to turn the IP compressor rotor stage 1 blades as necessary.

This will turn the IP system.

1. Remove the Turning tool (HU43985-2) carefully from the LP compressor blades, the engine section stator and the VIGVs.



IP TURNING TOOL

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

HPC Turing, Job Set-Up

Aircraft maintenance configuration –

TRENTXWB-A-72-31-13-00A01-722C-D.

Safety precautions –

TRENTXWB-A-72-31-13-00A01-722C-D

Make sure that the engine 1, (2) shutdown occurred not less than 5 minutes before you do this procedure.

Open the applicable circuit breaker(s):

TRENTXWB-A-72-31-13-00A01-722C-D

Getting access.

Open the fan cowls.

Put the applicable access platform into a safe position at the left side of the engine.

Put the applicable access platform into a safe position at the right side of the engine.

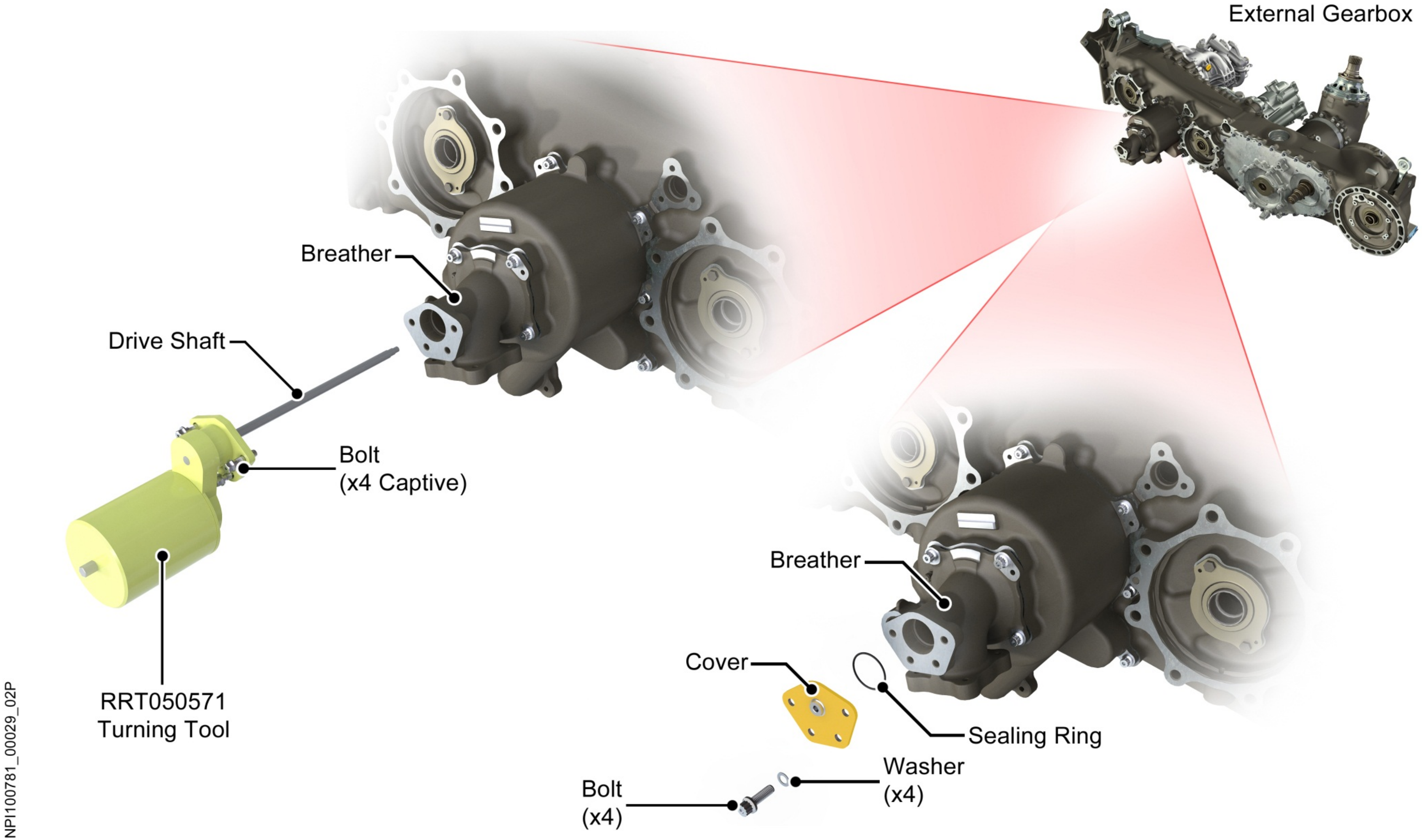
Procedure

1. Install the Immobiliser (RRT061241);
2. Refer to TRENTXWB-A-72-31-13-00A01-722C-D.
3. Install the Turning tool (RRT050571).
4. Hold the cover and remove the four bolts and the four washers.
5. Remove the cover from the External gearbox remove and discard the sealing ring.
6. Install the Turning tool (RRT050571) to the breather.
7. Ensure the drive shaft has engaged with the breather.
8. Install the four screws to attach the Turning tool (RRT050571) to the External gearbox.
9. Use an applicable wrench to turn the Turning tool (RRT050571). This will turn the HP system.

Note

If the tool clutch disengages, remove the tool; find the cause before you continue.

1. Remove the Turning tool (RRT050571).
2. Remove the four screws that attach the Turning tool RRT050571 to the External gearbox
3. Remove the turning tool from the breather.
4. Install a new sealing ring to the breather;
5. Refer to ROLLSROYCE-STDP-70-02-01-00A01-950A-A.
6. Lubricate the four bolts;
7. Refer to ROLLSROYCESTDP-70-70-03-00A01-950A-A.
8. Put the cover in its position on the External gearbox and install the four bolts and the four washers.
9. Use the Torque wrench No specific to torque the four bolts to the AMP torque figure
10. Refer to ROLLSROYCE-STDP-70-70-03-00A01-950A-A.
11. Remove the Immobiliser RRT061241;
12. Refer to TRENTXWB-A-72-31-13-00A01-522C-D.



HP TURNING TOOL

Maintenance Practices

Spinner Removal Procedure

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

AMP task TRENTXWB-A-72-21-11-00A01-520A-A.

The spinner position is identified for the number one position by an asterisk (*), this will ensure correct alignment on installation.

Remove the MORTORQ countersunk screws securing the spinner to the support ring.

Remove the nose cone spinner from the support ring.

NOTE

The nose cone spinner weighs 35 lbs. (16kg). Remove the 22 bolts from the spinner support ring. Using extractor tool (RRT065392) pull support ring clear of the LP compressor disc. Store the support ring a safe clean location.

Remove the 20 securing support ring bolts.

Fit jacking bolts into the spinner support ring.

Use the jacking bolts to release the spinner support ring and remove.

Annulus Filler Removal Procedure

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

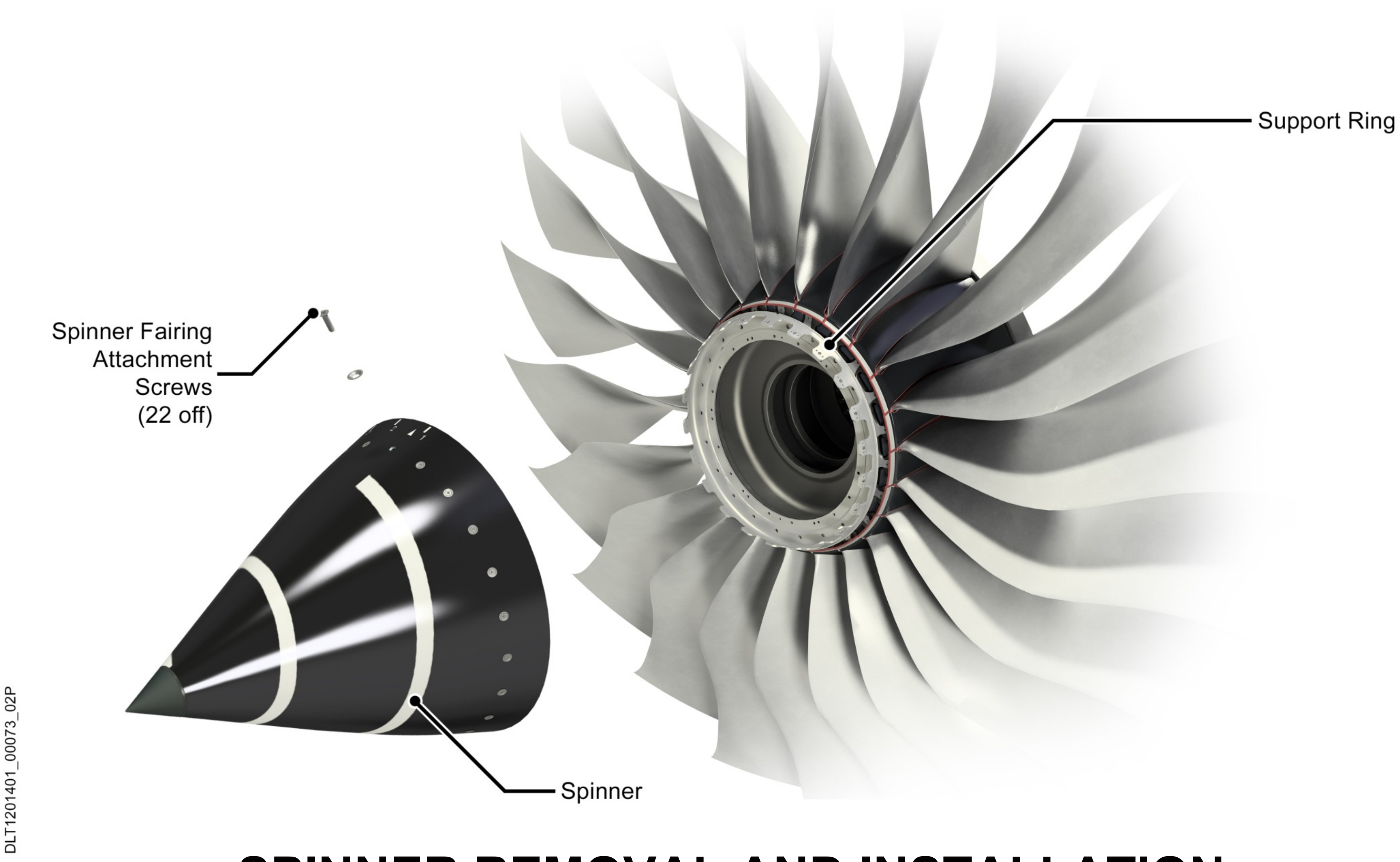
AMP Task TRENTXWB-A-72-31-13-00A01-520A-A

1. **NOTE**

   Using a temporary marker identify the location of each annulus filler with reference to the number one blade, which is identified by ←1 engraved on the disc face. This is to ensure they are re-installed in the same position.

2. To remove the annulus filler, pull the annulus fillers forward to disengage the hooks from the LP compressor disc, and then turn the annulus filler in the direction of its curve to clear the fan blade.
3. Remove all annulus fillers and store in a safe clean location.
4. **NOTE**

   The information on the annulus filler including serial, part number and weight is found on the underside at the rear of the filler.



SPINNER REMOVAL AND INSTALLATION

Maintenance Practices

Fan Blade Removal

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

AMP Task TRENTXWB-A-72-31-13-00A01-520A-A

WARNING

YOU MUST USE THE APPLICABLE GLOVES WHEN YOU HOLD THE FAN BLADES. THE SHARP LEADING EDGE OF THE BLADES CAN CAUSE CUTS TO HANDS.

WARNING

YOU MUST MAKE SURE YOU CAN HOLD THE WEIGHT OF THE BLADE BEFORE YOU REMOVE IT. IT IS HEAVY AND CAN CAUSE INJURY TO PERSONS AND DAMAGE EQUIPMENT.

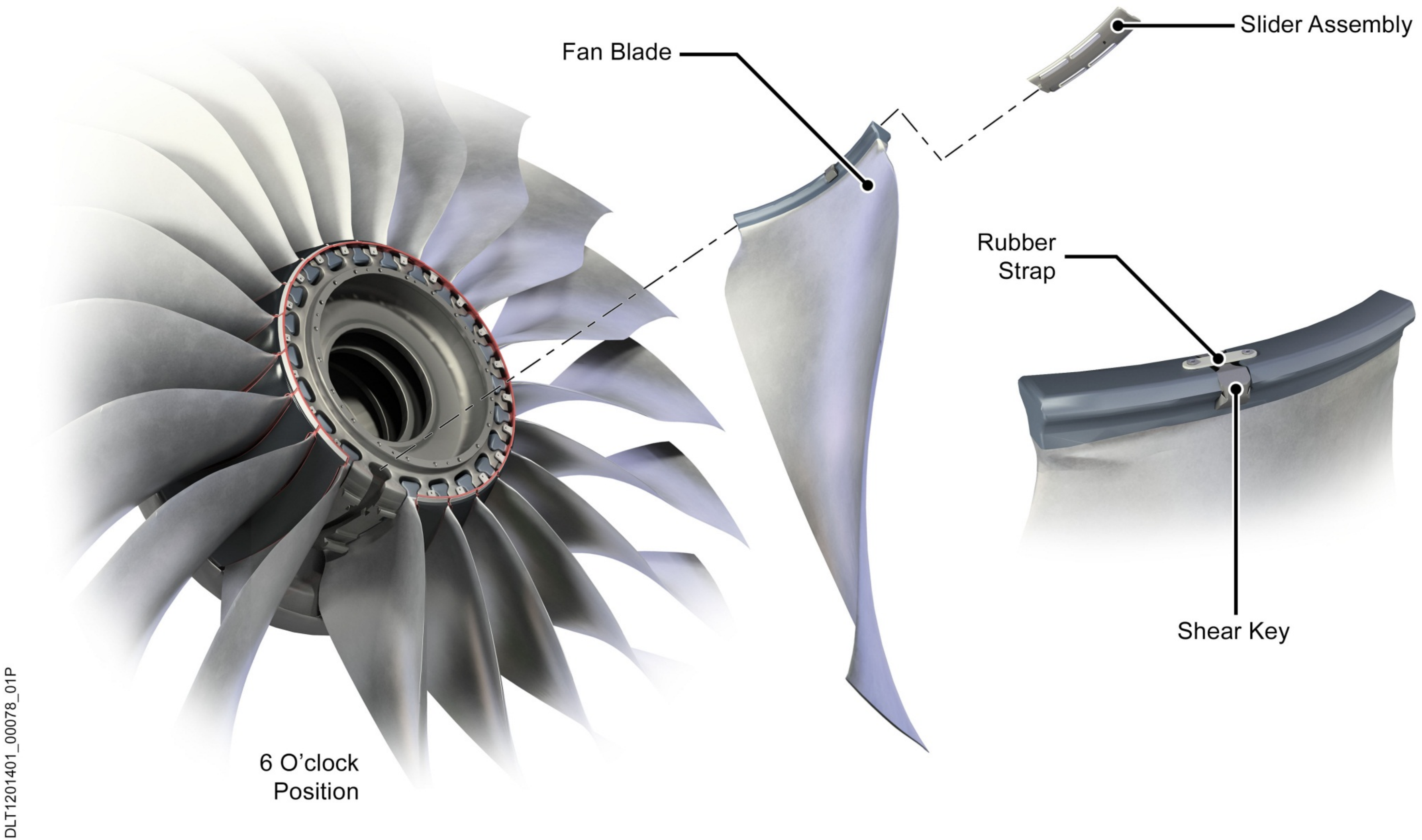
CAUTION

YOU MUST MAKE SURE THE BLADES DO NOT TOUCH ADJACENT BLADES WHEN YOU REMOVE IT. IF YOU LET THE BLADES TOUCH EACH OTHER, YOU CAN CAUSE DAMAGE TO THE BLADES.

Note:

The LP compressor blades weigh approximately 40 lbs. (18.2 kg).

1. Use an approved temporary marker to identify all 22 blade positions in an anti-clockwise direction with reference to the number 1 blade, which is identified by ←1 engraved on the disc face.
2. Turn the LP rotor so that the No1 blade is at the bottom dead centre (BDC) and install immobilizer (RRT061241) to prevent movement of the fan assembly.
3. Using extractor (HU29255) & adapter (HU44819) remove the slider assembly from the dovetail slot. Identify the slider with a temporary marker Lift the blade to disengage the shear key then carefully pull the blade forward to remove it.
4. Record the part number (P/N), serial number (S/N) of the blade in an appropriate document to assist the installation procedure.
5. Remove the immobilizer (RRT061241) and carefully turn the LP rotor until number 12 blade is at the BDC position. Remove the number 12 blade.
6. Repeat the procedure for the remaining blades in this sequence No13, No2, No3, No14, No15 and No4 etc. until all 22 blades are removed.



FAN BLADE REMOVAL

Maintenance Practices

Fan Trim Balance

The Trent XWB engine has 22 fan blades and a dedicated flange of 22 holes in the LP support ring, this allows fitment of the Fan Trim Balance weights as follows;

- No weight
- Small trim balance weighted bolt (BLT5682)
- Medium trim balance weighted bolt (BLT5679)
- Large trim balance weighted bolt (FW24476)

The spinner support ring also houses two further flanges a mid-ring flange consisting of 20 holes, and an inner ring flange of 80 holes.

In certain circumstances if directed to by the AMP the 20 bolt flange can be used for LP balancing.

Under no circumstances is the 80 bolt flange to be used for LP balancing, this is only used for module balancing only at manufacture.

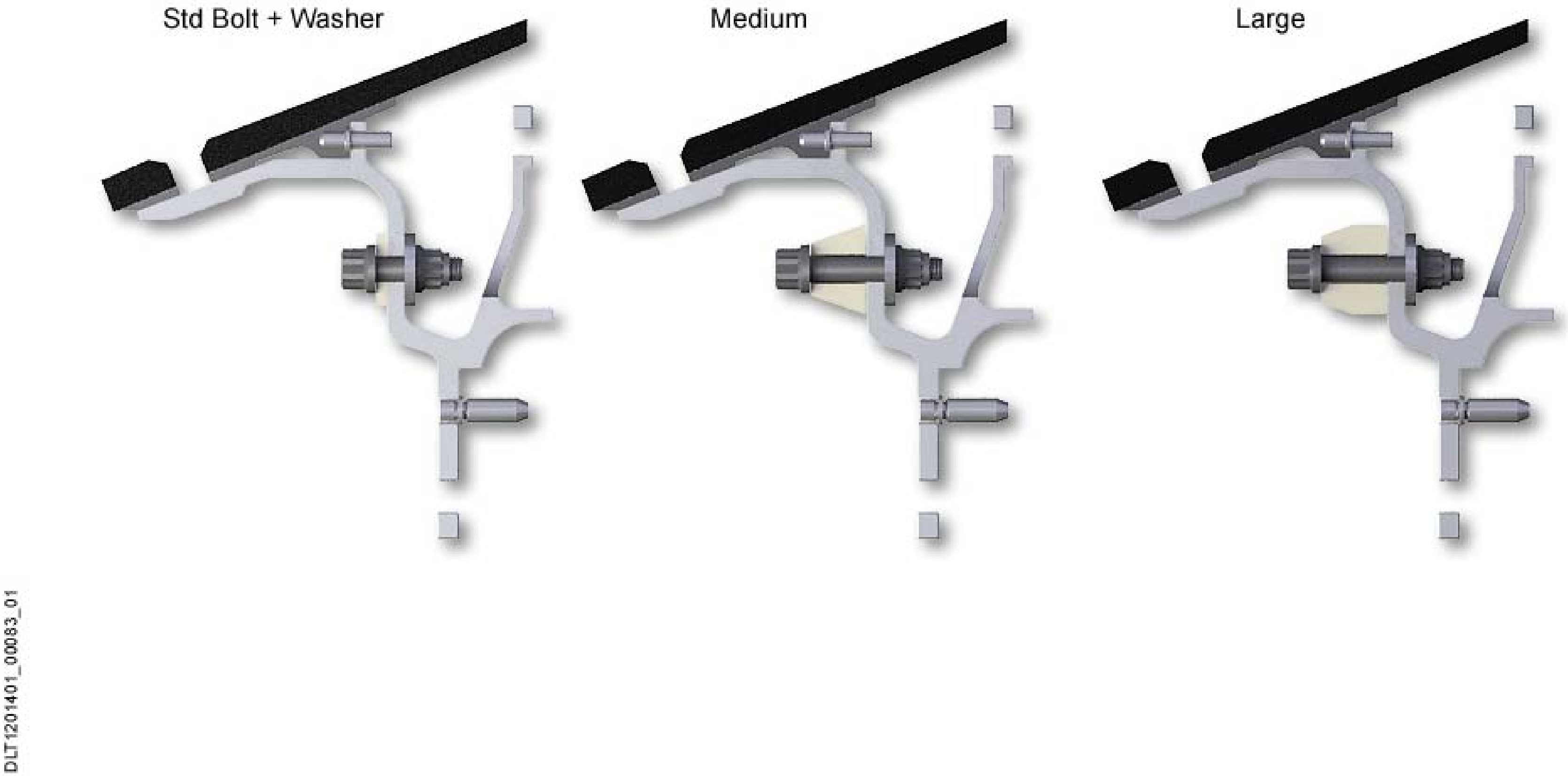
The Trent XWB engine will use a one-shot LP Trim Balance system.

The basis of the One Shot LP Trim Balance system is that, using data from the previous engine run, along with some stored Influence Coefficients, an LP trim balance solution can be calculated without the need for additional engine runs. This balance solution can then be fitted and, if necessary, an engine check run performed to ensure that the solution is adequate and has been properly fitted.

Influence Coefficients are used to convert the engine response levels at the Aircraft Vibration Monitoring (AVM) position into levels of imbalance on the LP Rotor. They are calculated for each LP Speed, typically from 50%NL to 100%NL and are calculated from engine runs with known different levels of LP imbalance. Influence Coefficients that are calculated by an Engine Monitoring unit (EMU) for a specific engine are referred to as Specific Influence Coefficients (SIC); Influence Coefficients that are effectively an average of several sets of SIC are referred to as Generic Influence Coefficients (GIC).

For service operation it is intended that GIC will be already loaded within the EMU in order that a one-shot LP trim balance can be performed at any time, but if for a particular engine the GIC do not allow a suitable Trim Balance solution to be calculated, then additional runs with known additional imbalances applied will need to be run in order that the SIC can be calculated.

These SIC will then be the default option for any future Trim Balance calculations for that engine installed on that aircraft. The initial values for the GIC will be based on the SIC deduced from the A350 Flight Test Program.

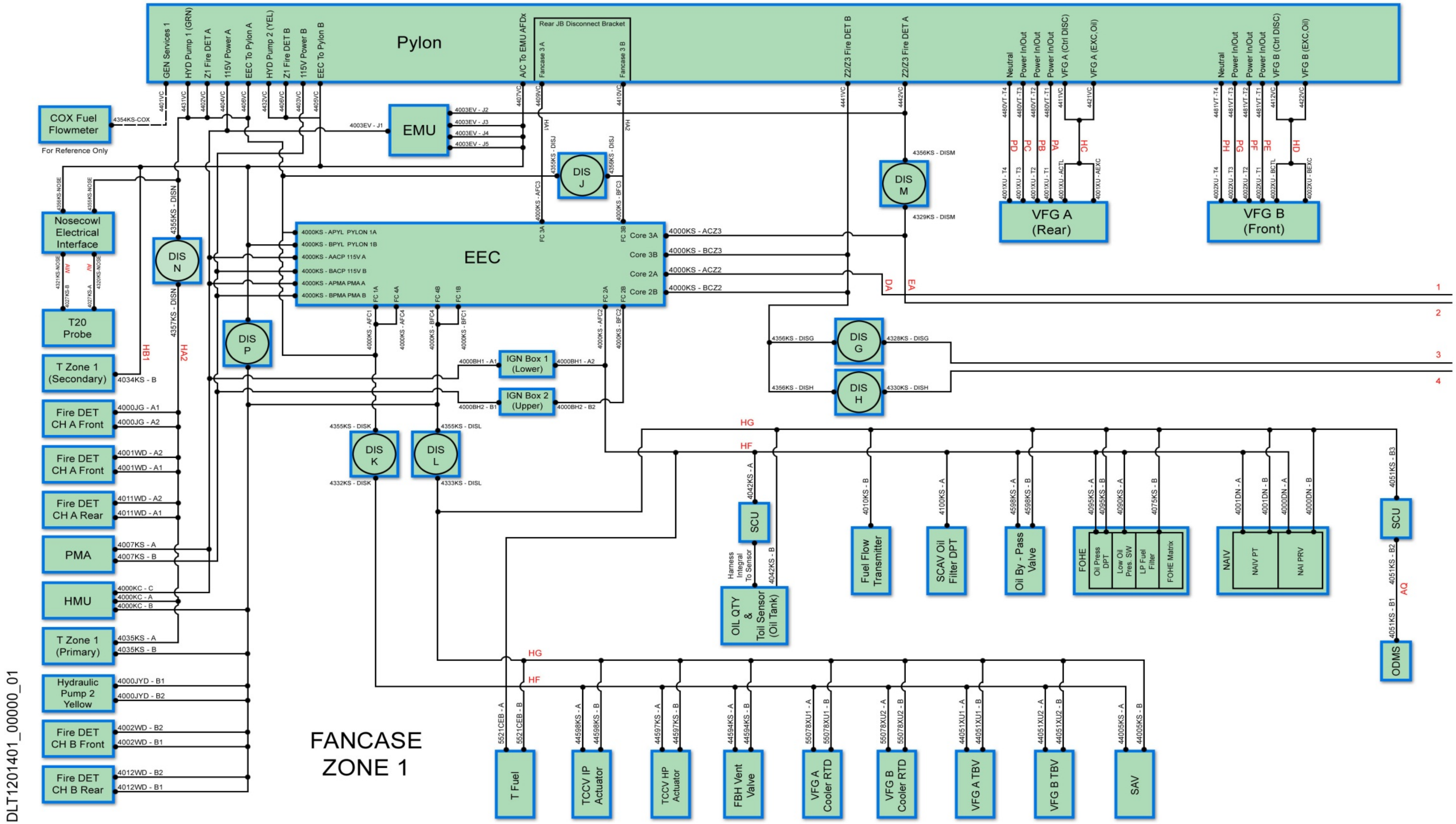


FAN TRIM BALANCE

Mechanical Arrangement Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.



ZONE 1 WIRING DIAGRAM

Section 4 – The Trent XWB Engine

Objectives

The trainee should now be able to:

- State the bearing arrangement of the Trent XWB engine.
- Recognise the modular breakdown of the Trent XWB engine.
- Identify the location and describe the purpose and operation of the engine modules.
- Identify and locate the Trent XWB engine components installed on the left and right side of the engine.
- Identify the location and describe the purpose of the borescope access positions on the Trent XWB engine.
- Recognise the reasons and components removed for separating the Trent XWB for transportation purposes.

End of the Trent XWB Engine Section

Section 5 – Propulsion Control System (PCS)

Section 5 – Propulsion Control Systems

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance documentation level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Propulsion Control System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Propulsion Control System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Propulsion Control System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Propulsion Control System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Propulsion Control System interfaces with other engine and aircraft systems.

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Propulsion Control System (PCS)

Introduction

The Airbus A350 / Trent XWB Propulsion Control System (PCS) is a combination of interfaces between the engine mounted Engine Electronic Controller (EEC), Engine Monitoring Unit (EMU), airframe mounted Engine Interface Function (EIF) and Electronic Thrust Reverser Actuation Controller (ETRAC).

The Heart of the Engines FADEC system is the EEC, which is a digital electronic computer based control unit that interfaces with all sensors, actuators and communication links that make up the FADEC. This system provides all the functions needed to start, run and shutdown the engine, to communicate with the Engine Health Monitoring (EHM) system, the aircraft for cockpit indications, and to provide maintenance information.

In order to carry out its function the FADEC employs numerous devices to monitor the engine / aircraft. The EEC processes the data it receives from these devices and outputs appropriate control and condition signals to the engine / aircraft and also supplies relevant information to the EHM system and the cockpit.

The FADEC system must carry out its function while maintaining a safe and efficient engine condition throughout the entire operating envelope of the engine.

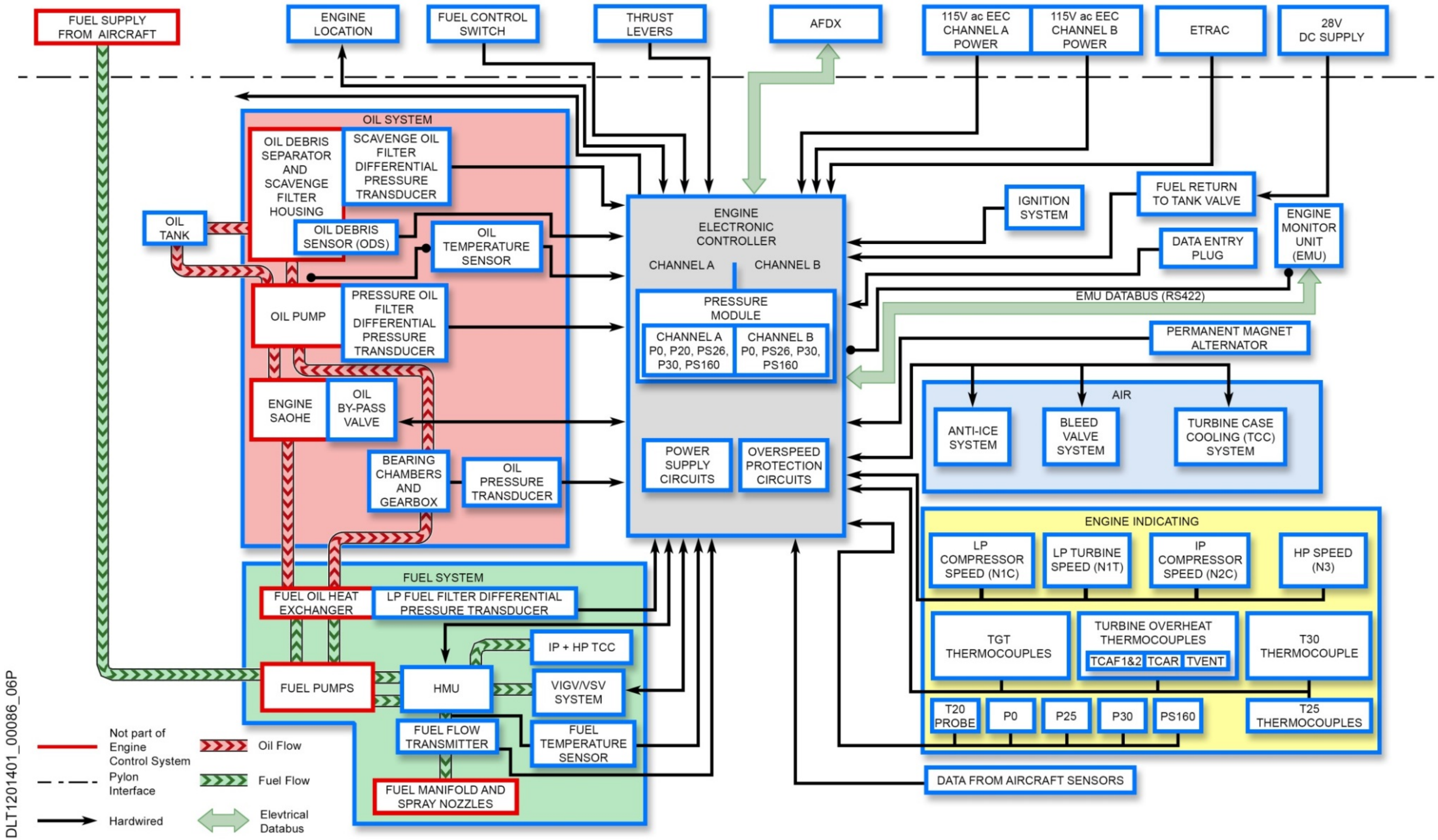
The flight crew retain ultimate authority over the engine via the Fuel Master Start ON / OFF switch, which is hardwired directly to the Shut-off Servo Valve in the Hydro-Mechanical Unit (HMU). Selecting the switch to the OFF position overrides any PCS commands and shuts the engine down.

The FADEC system provides a degree of fault tolerance by using a dual channel system. If a fault occurs that affects correct functioning, the other channel can take control of the function, minimising the risk of faults developing into a system or engine failure, and at the same time, maximising engine / aircraft availability.

FADEC Functions

The following functions are provided by the FADEC

- FADEC Power supplies
- Data Entry Plug
- Engine Protection
- Engine Health Monitoring (EHM)
- Cockpit Indications (Section 6)
- Heat Management (Sections 7 & 8)
- Thrust Management (Section 8)
- Fuel & Control (Section 8)
- Airflow Control (Section 9)
- Engine Ventilation and Cooling (Section 10)
- Ice Protection (Section 12)
- Starting & Ignition (Section 13)
- Maintenance system.



FADEC OVERVIEW

Airframe / PCS Interfaces

The majority of the airframe mounted computer systems are installed in the forward Avionics Compartment. The interfaces between these systems LRUs are mainly passed through the Integrated Modular Avionics (IMA) communications network, which is also known as the Avionics Full Duplex Ethernet (AFDX) network. Any LRUs not directly connected to the AFDX network; connect to the network via Core Processing Input / Output Modules (CPIOMs) or Common Remote Data Concentrators (CRDCs).

Avionics Full Duplex Ethernet Network (AFDX)

The AFDX can be likened to a company's intranet that forms a digital communication network for the Airbus A350. Data is placed on the AFDX network for the intended recipient to retrieve. The Trent XWB PCS communicates with the aircraft via the AFDX network.

Controller Area Network (CAN) Data Bus

The CAN data bus is a digital communications bus used as a secondary back-up link between the ECS and aircraft, in case of total loss of the AFDX communications link. This data bus has limited capacity compared to the AFDX and therefore can only support critical parameters such as Air Data.

Air Data

Inputs, from the aircraft and engine, provide independent

- Static Air Pressure – aircraft PStatic & engine P0.
- Total Air Pressure – aircraft PTotal & engine P20.
- Total Air Temperature – aircraft TAT & engine T20.

The Airbus A350 has a single platinum Resistive Temperature Device (RTD) T20 TAT probe located in each engine intake. The T20 sensing probe is not heated. The engine does not have a Total Air Pressure sensor (P20) in the air intake, four independent airframe mounted sources are used and the aircraft selects a Total Pressure reading parameter from these sources for use by the EEC. P0 Static Air Pressure is measured via vents in the EEC casing. Aircraft air data values are transmitted to the PCS via the AFDX or CAN.

Core Processing Input / Output Modules (CPIOM)

The CPIOMs host avionics applications and process data to execute avionics functions. One CPIOM hosts a number of applications. The CPIOMs convert and transmit data between the Avionics Data Communication Network (ADCN) and LRUs that do not have the AFDX technology.

Common Remote Data Concentrator (CRDC)

CRDCs collect, convert and exchange data between the ADCN and LRUs that do not have the AFDX technology and that are mostly installed out of the avionics compartment.

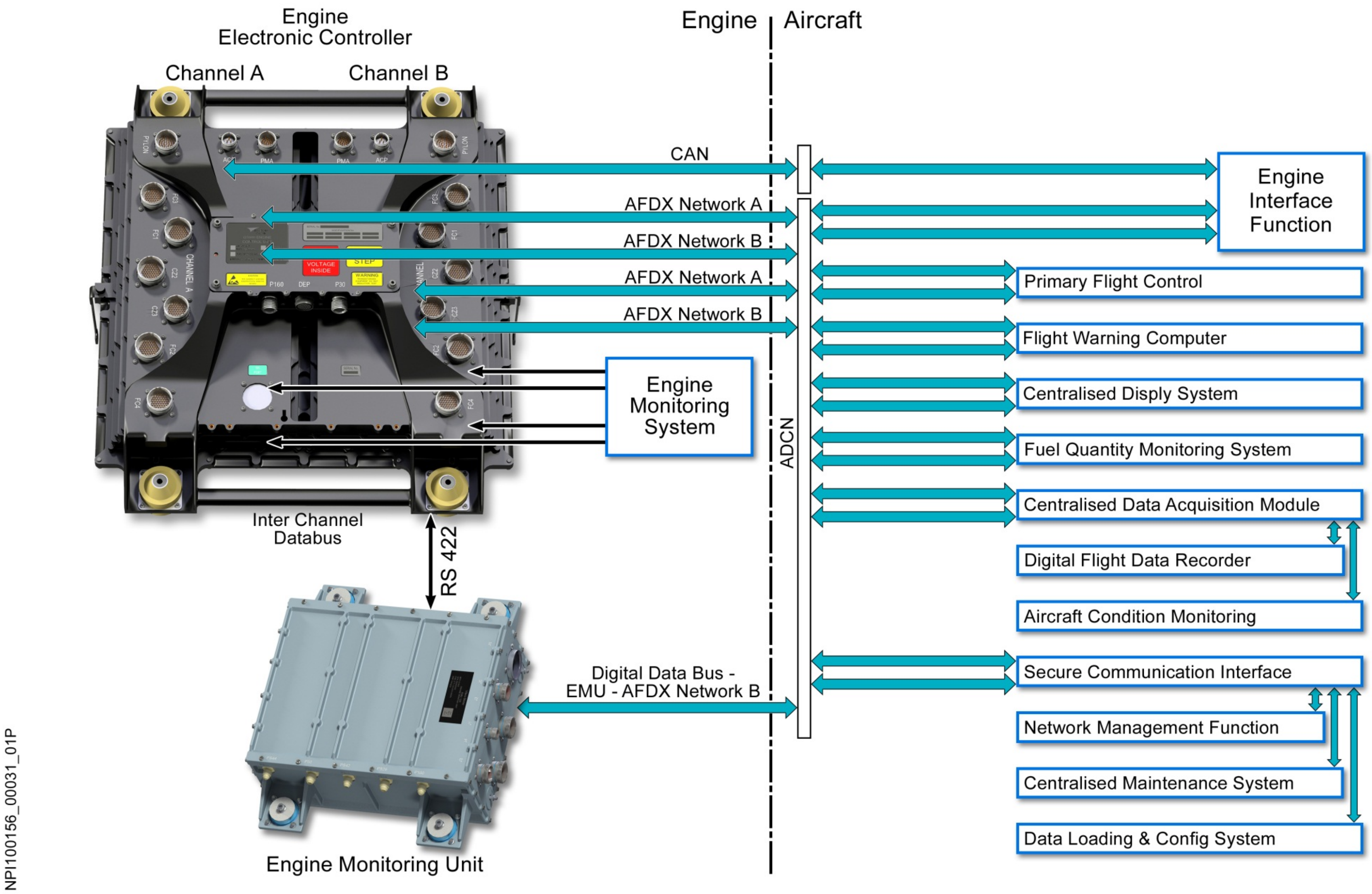
Engine Interface Function (EIF)

The EEC and EIF are the heart of the PCS. They work together as a team and share the engine primary functions. The EIF receives cockpit selection and sends data to the EEC for engine starting and shutdown.

The EIF sends the throttle- level position signals and the air data to the EEC for thrust management.

The EIF controls the supply of electrical power to the EEC, ETRAC & EMU from the aircraft electrical system.

The EIF is the primary interface between the EEC and the aircraft systems. The EIF receives, selects and consolidates numerous aircraft data to send to the EEC and also consolidates engine data to send to different aircraft systems.



AIRFRAME / EEC INTERFACE

EEC Electrical Power Configuration Overview

The aircraft supplies single-phase 115V AC variable frequency electrical power independently to each channel of the EEC. An 115V AC emergency power bus provides power to channel A and an 115V AC normal bus provides power to channel B.

Normally both power supplies are available when full aircraft power is in operation. The emergency power supply is backed up by battery and will be available in conditions when only limited power is available within the aircraft (in these conditions the aircraft normal bus will not be available).

The EEC will use the aircraft power when the engine HP shaft speed is below 6.46% and the PMA cannot provide sufficient power to operate the EEC. This will be the case when the engine is shut down and static on the ground and shut down in-flight and windmilling below 6.46%. The aircraft power will also be selected if the EEC is being powered by the PMA and the PMA power to the EEC fails.

During fault free operation, engine control and transmission of data to the aircraft is not affected by loss or interruption of either or both aircraft power sources to the EEC. This is because the engine dedicated power source the PMA powers the EEC.

Loss of PMA power to the controlling EEC channel will result in a channel change to the other channel (which has PMA power available). A total loss of PMA power to the EEC results in the EEC being dependant on aircraft power. In this case, loss or interruption of the aircraft power to the controlling EEC channel will result in a channel change to the other channel. The change of channel in control in these circumstances will ensure no hazardous effect on engine control and no transmission of erroneous data to the aircraft.

Engine Monitoring Unit (EMU) Power

The EMU is powered by 115V AC power from the aircrafts L (R) Main AC Bus.

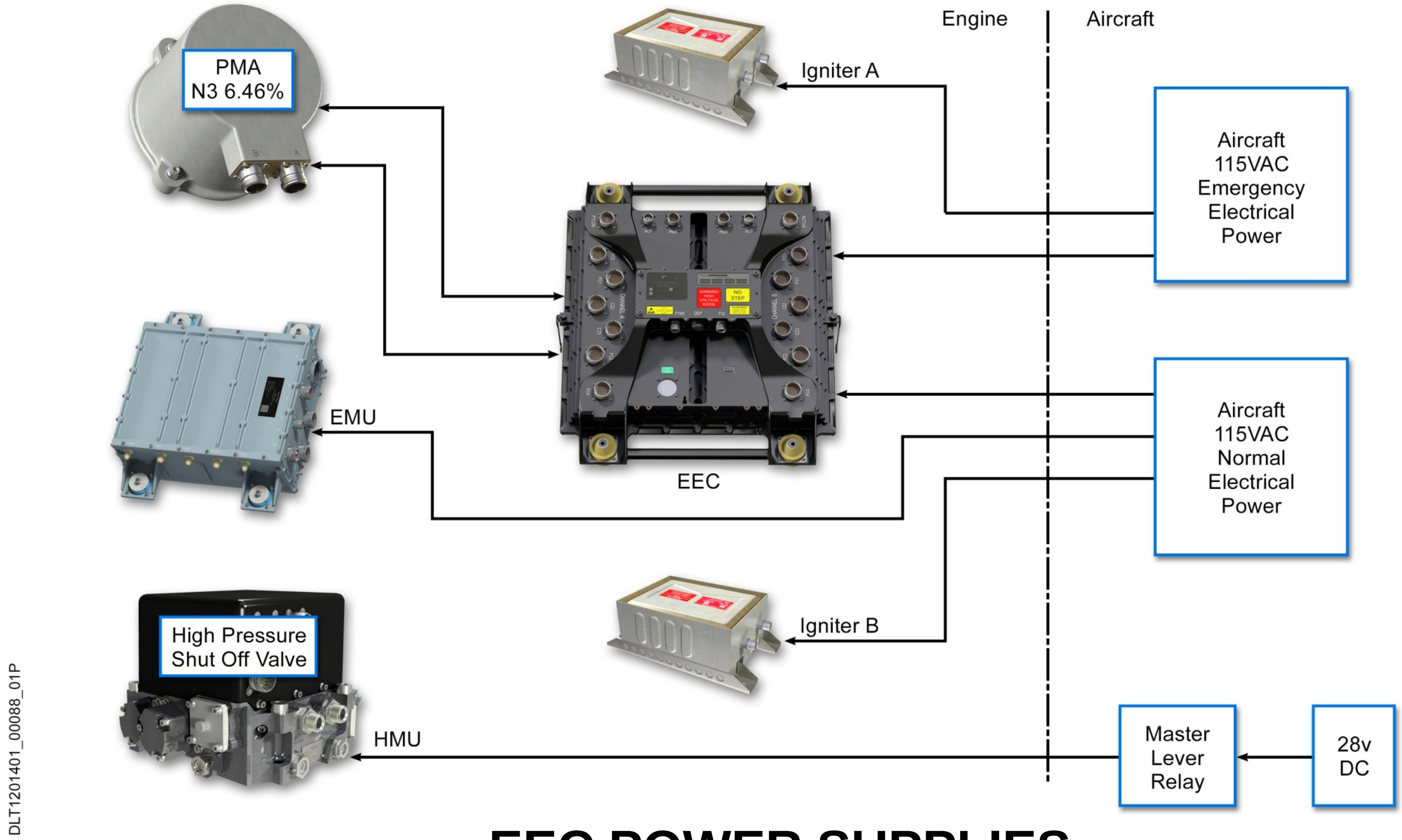
Ignition System 1 & 2

Aircraft 115V AC is used to energise the ignition systems, when the Master Lever is placed in the ON position.

28V DC Supplies

Pressure Raising and Shut Off Valve (PRSOV)

Aircraft 28V DC is used to energise the PRSOVs shut-off torque motor windings closed, and shutdown the engine when the ON / OFF Master Lever is moved to the OFF position.



EEC POWER SUPPLIES

Permanent Magnet Alternator (PMA)

Location

The PMA is located on the External Gearbox front face left side.

Purpose

To provide the main source of electrical power for the EEC when the engine is running and to provide a HP shaft (N3) speed signal to the:

- Aircraft
- EEC
- EMU

Description

The PMA consists of two separate Line Replaceable Units (LRUs) – a Stator and Rotor. They are not supplied as matched pairs.

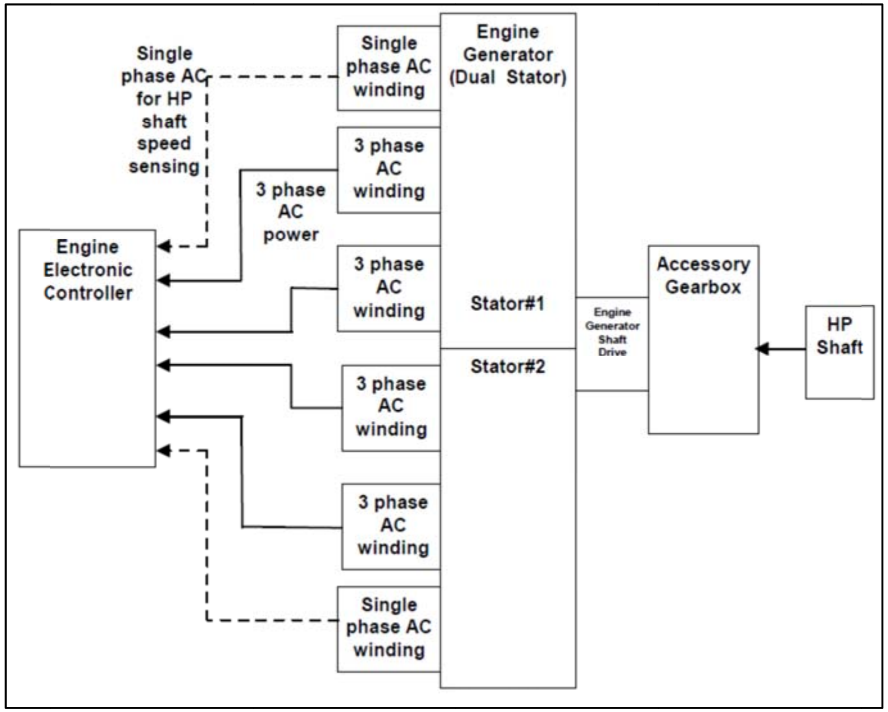
The PMA has two independent 3-phase windings, each providing a separate source of AC power to their respective EEC channels.

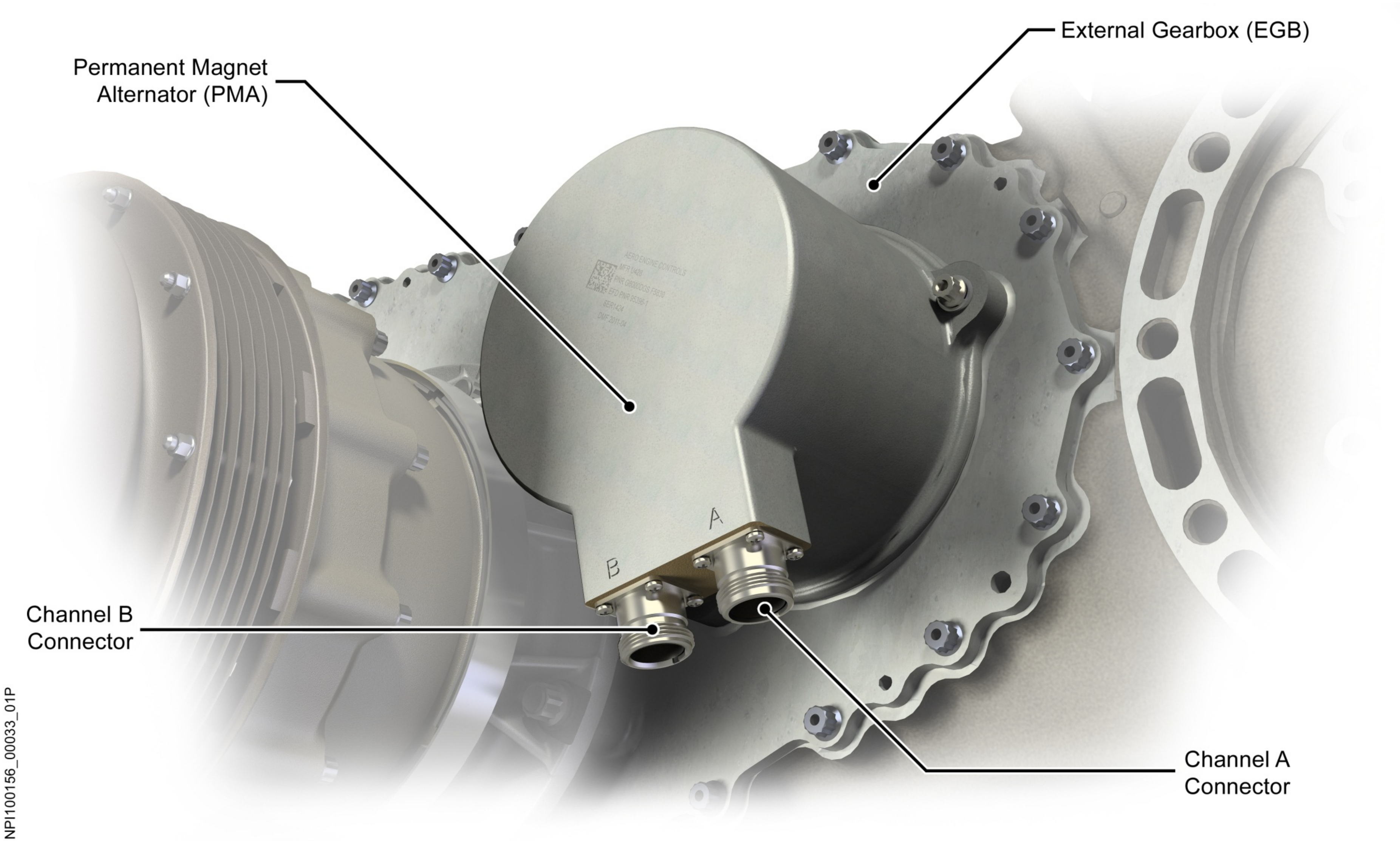
At engine speeds from 0 to 6.46% N3, the output from the PMA is insufficient to power the EEC, and aircraft power is used instead. Once N3 speed is greater than 6.46% the PMA is used to supply power to the EEC.

A separate single-phase winding on each channel within the PMA provides primary N3 (HP shaft) speed signals to Channel ‘A’ and Channel ‘B’ of the EEC.

Within the EEC, N3 signals are additionally buffered to supply the EMU with HP shaft speed data for engine health monitoring and speed recording.

An N3 speed signal is sent to the aircraft independent of the EEC for back up.





PERMANENT MAGNET ALTERNATOR

Engine Electronic Controller (EEC)

Location

The EEC is located on the left side of the engine fan case approximately at the 10 o' clock position, and is secured by bolts fitted through four anti-vibration mounts.

Purpose

The EEC is the central control unit of the FADEC. It is a dual channel computer system that interacts with a series of engine and aircraft interfaces to manage engine starting & shutdown, thrust setting and engine protection.

EEC Architecture

An array of engine and aircraft sensors sends information to two virtually identical, but independent electronic circuits, inside the EEC called Channel A & Channel B.

Both channels have their own separate power supplies, and can manage all the functions needed for engine control independently.

In usual operation both channels are powered, but only one channel is "in control" at any one time. Each channel will be forced into control every other power up, providing that both channels health score is equal.

Each channel contains three computers, a Control Computer, a Protection Computer and a Communication Computer. The Control Computer performs the engine control functions and the Protection Computer independently monitors for hazards and shuts the engine down if a major hazard is detected. The Protection Computers of both channels monitor for hazards even when their respective channel is not in control.

The Communication Computer is the interface between the engine and the aircraft using the AFDX / CAN

Power Supply and Selection

Each channel of the EEC receives its own power supply from the PMA and aircraft 115V AC. Both these sources of power are converted internally into a 45V DC voltage. The EEC selects power from whichever source gives the highest voltage to form the Intermediate Power Rail (IPR), which is nominally 45v DC.

The PMA derived voltage is deliberately set higher than aircraft derived voltage so that when the PMA voltage is available it is always used.

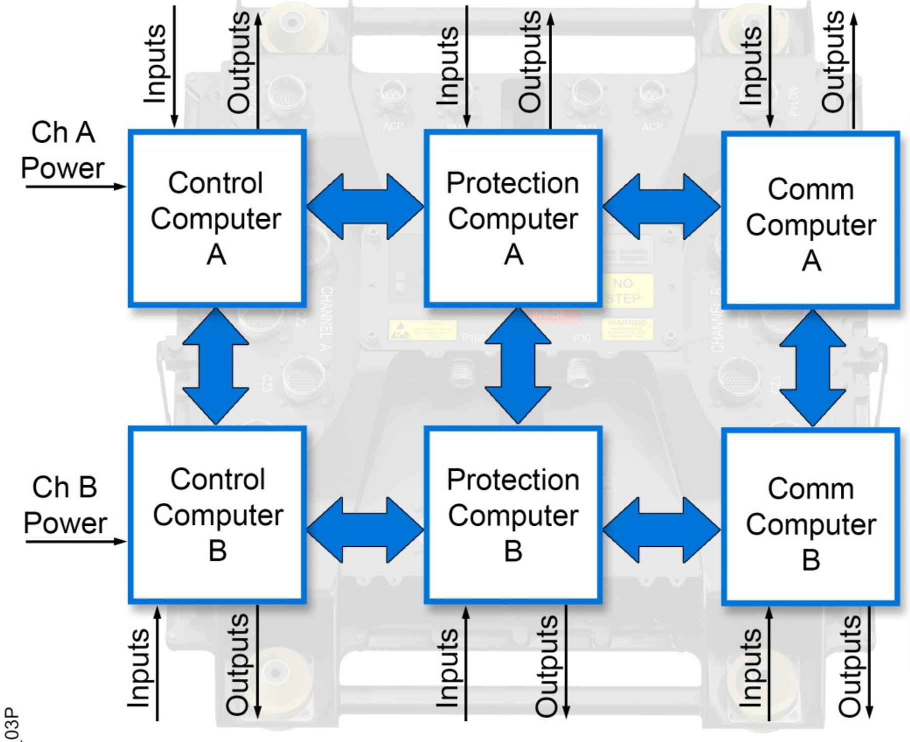
Inputs and Outputs

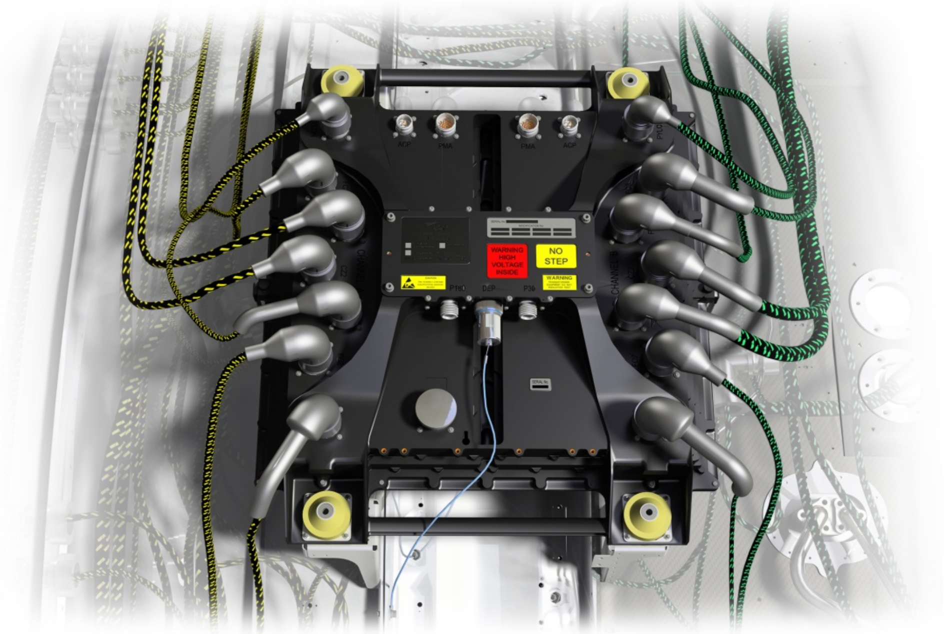
There are two main types of sensor input from the engine to the EEC:

- Control sensors – provide signals needed for closed loop control of the engine.
- Condition sensors – indicate engine conditions that do not directly affect closed loop control.

The EEC uses the inputs it receives to generate controlling outputs to the torque motors and solenoids of the various systems that interact with the EEC. The EEC also outputs condition and status information to the EHM system and the aircraft.

A dual channel Data Entry Plug (DEP) is physically plugged into a connector at the bottom of the EEC to provide the unique engine data to the EEC.





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EEC ARCHITECTURE

EEC Electrical Connectors

Introduction

The EEC is split into two distinct sections that reflect the two channels of control. The connectors on the left are dedicated to channel A, and the connectors on the right to channel B. At the base of the unit are the sensed pressure pipe connections together with the Data Entry Plug.

Purpose

The EEC measures, processes and activates a series of engine and aircraft interfaces for the purpose of safely controlling the engine. It is the central control unit of the FADEC system, which comprises of sensors, actuators, power supplies, data buses and interconnecting harnesses. The harnesses must all be in good condition to maintain efficient engine control over the whole range of operation. For the purposes of understanding and troubleshooting, the connectors are identified on the EEC casing and on the harness identifying tags.

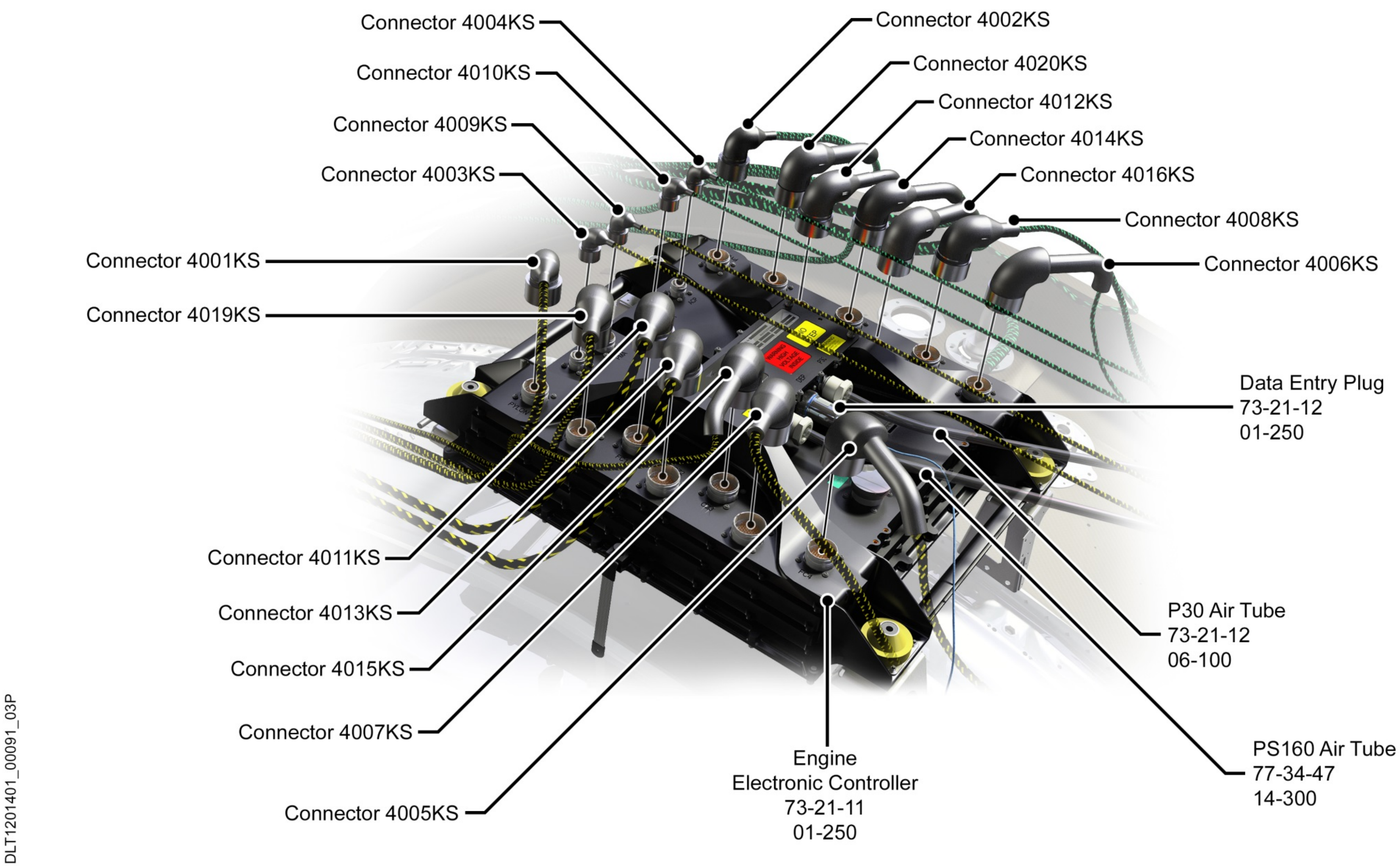
FC = Fan Case

Pylon = Engine / Airframe interface pylon

AFP = Airframe Power supplies

PMA = Permanent Magnet Alternator

CZ = Core Zone



EEC ELECTRICAL CONNECTORS

Engine Over-Threat Protection Systems

The Trent XWB EEC has a five protection systems, each designed to detect and react against a range of different threats, which if not otherwise accommodated, may develop into a more hazardous event.

N1 Turbine Overspeed System (LPTOS)

Introduction

In the event of a break in the LP turbine shaft, the turbine would be freed from its role of driving the LP compressor. If ignored such an event would lead to a turbine overspeed with a potential for the release of high-energy debris outside the confines of the engine.

Purpose

Rapid detection and accommodation of an LPT overspeed.

Description

If an LPT overspeed is detected the N1 turbine overspeed function will automatically:

- Stop fuel flow to the engine.
- Open all bleed valves.
- Move the VSVs to the closed (low speed) position.

These actions minimise the peak overspeed, thereby reducing the likelihood of an uncontained turbine failure.

An N1 turbine overspeed is detected by comparing measured LP compressor and turbine speeds.

These measurements are obtained from 4 LP compressor speed probes and 4 LP turbine speed probes. The speed probe signals are sent to the EEC Protection Computers as follows:

- Two LP Compressor speed probes (N1C-1 and C-2) and two LP Turbine speed probes (N1T-1 and T-2) are compared in Channel A.
- Two LP Compressor speed probes (N1C-3 and C-4) and two LP Turbine speed probes (N1T-3 and T-4) are compared in Channel B.

N1 Shaft Breakage Detection

Either channel's Protection Computers can independently detect a shaft breakage and if deemed valid will:

- Energise the PRSOV torque motor in the HMU to shut off the fuel flow within 4 ms of the breakage being detected.

The channel in control 'Control Computer' will then:

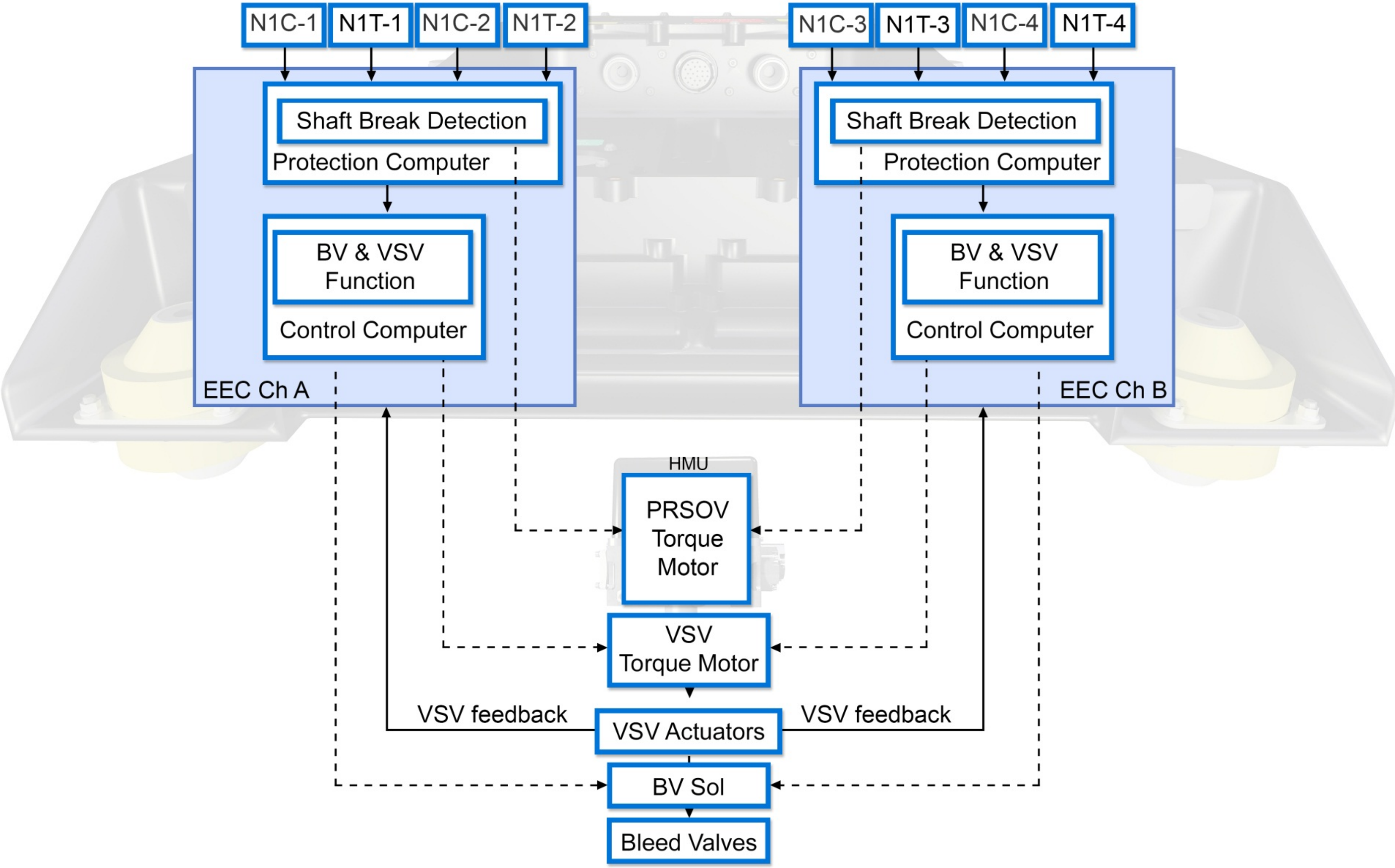
- Open all the handling bleed valves.
- Close the VSVs i.e. moved to the low speed stop.

The function is then latched until the flight crew move the Master Lever from the ON to the OFF state.

Signal loss

When one detector is lost (due to a single speed probe failure) and the other EEC channel has both detectors available then LP shaft break detection in the faulty channel is inhibited. Both detectors on the non-faulty channel must agree to trigger engine shutdown.

When one detector has lost on each EEC channel then the system reactivates the working detector on each channel, and each detector can independently trigger an engine shutdown.



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LP TURBINE OVERSPEED (LPTOS)

Engine Over-Threat Protection Systems

N2 Turbine Overspeed System (IPTOS)

Introduction

In the event of a break in the IP turbine shaft the turbine would be released from its role of driving the IP compressor. If ignored such an event would lead to a turbine overspeed with a potential for the release of high energy debris outside the confines of the engine.

Purpose

To provide rapid detection and accommodation of an IPT overspeed.

Description

If an IPT overspeed is detected the IP turbine overspeed function will automatically:

- Stop fuel flow to the engine.
- Open all the HP & IP handling bleed valves.
- Move the Variable stator vanes to the low speed position.

The actions minimise the peak overspeed, thereby reducing the likelihood of an uncontained failure.

The EEC monitors inputs from Turbine Cooling Air Front (TCAF 1 & 2), Turbine Cooling Air Rear (TCAR), P30 and IP speed probes to detect the conditions under which an IP shaft failure may occur.

Shaft Breakage Detection

Either channel's Protection Computers can independently detect a shaft breakage and if deemed valid will:

- Energise the PRSOV torque motor in the HMU to shut off the fuel flow.

The channel in control 'Control Computer' will then:

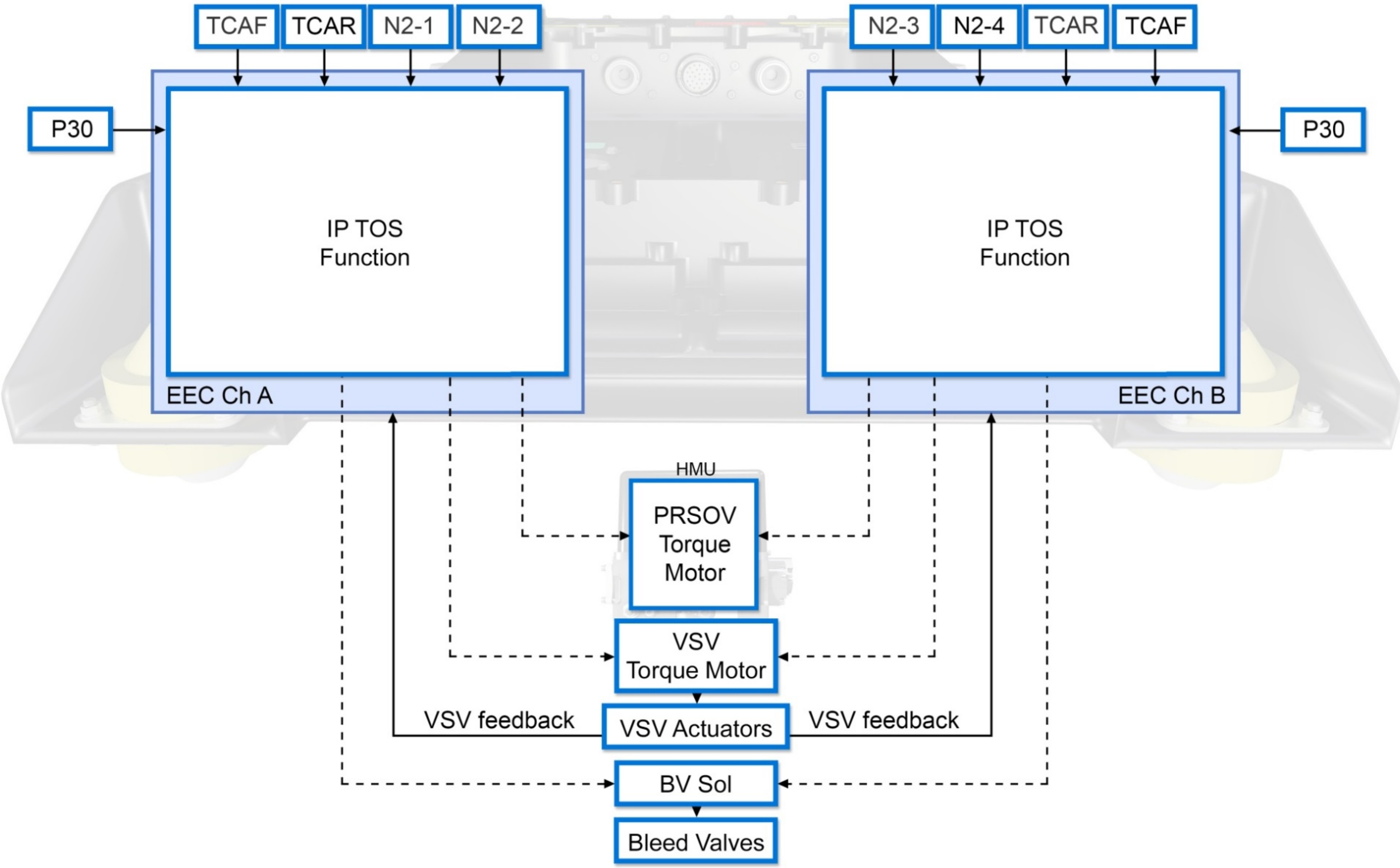
- Open all the handling bleed valves.
- Close the VSVs i.e. moved to the low speed stop.

The function is then latched until the flight crew move the Master Lever from the ON to the OFF position.

Signal loss

When only one valid IP shaft speed sensor reading is available or there is a crosscheck disagreement between the two sensors in a channel and the other EEC channel has both IP speed sensors available then IP shaft break detection in the faulty channel is inhibited.

When only one valid IP shaft speed sensor reading is available or there is a crosscheck disagreement between the two sensors in a channel and the other EEC channel also has a degraded IP speed configuration then IP shaft break detection is allowed using the single valid speed sensor.



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IP TURBINE OVERSPEED (IPTOS)

Engine Over-Threat Protection Systems

N1 System Overspeed Protection

Introduction

An overspeed of the LP Shaft may be caused by a malfunction of the control system to correctly regulate fuel flow.

Location

The N1 Shaft Overspeed function resides in each channel of the Protection Computer of the EEC.

Purpose

The purpose of the N1 shaft overspeed function is to detect an overspeed condition and minimise the potential release of high-energy debris by rapidly shutting off fuel flow to the Fuel Spray Nozzles.

Description

Both channels of the Protection Computers (A and B) are capable of detecting an LP (N1) overspeed event by comparing the compressor speeds to an N1 maximum threshold. The LP Turbine speeds are also used but as back-up only should one of the compressor speed probes fail.

The Protection Computers receive the following speed signals:

Channel A

Two N1 compressor signals (N1C-1 and N1C-2) and one N1 Turbine signal (N1T-2).

Channel B

Two N1 compressor signals [N1C-3 and N1C-4] and one N1 Turbine signal (N1T-4).

Either channel of the EEC can detect and initiate an overspeed event independently of the other channel, whether the channel is in control or not, providing two valid speed signals are available i.e.

- Compressor and Compressor or
- Compressor and Turbine.

Signal Loss

Either channel's Protection Computers can independently detect a shaft overspeed and if deemed valid will:

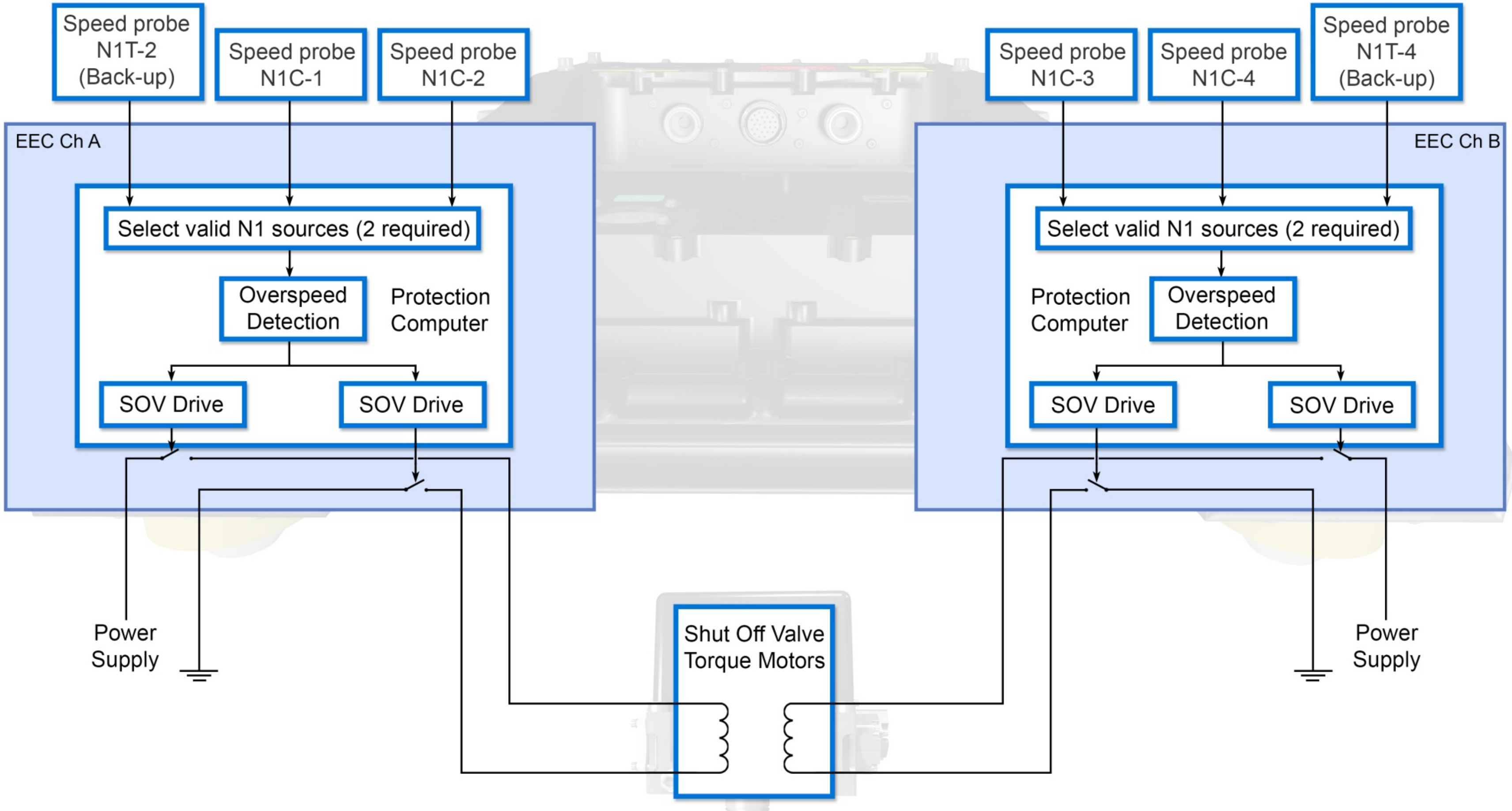
- Energise the PRSOV torque motor in the HMU to shut off the fuel flow.

The function is then latched until the flight crew move the Master Lever from the ON to the OFF state.

If two of the three available signals within a channel are invalid then the function in that channel is disabled but the other channel will still provide protection.

BITE Checks

The LP shaft overspeed system is subject to a BITE test to confirm correct system operation.



N1 SHAFT OVERSPEED

N2 System Overspeed Protection

Introduction

An overspeed of the IP Shaft may be caused by a malfunction of the control system to correctly regulate fuel flow.

Location

The N2 shaft overspeed function resides in the Protection Computers of the EEC.

Purpose

The purpose of the N2 shaft overspeed system to detect an overspeed condition of the IP compressor and minimise the potential release of high-energy debris by rapidly shutting off fuel flow to the Fuel Spray Nozzles.

Description

Both channels of the Protection Computers (A and B) are capable of detecting an IP (N2) overspeed event by comparing the compressor speeds to an N2 maximum threshold.

The Protection Computers receive the following speed signals:

Channel A

Two N2 Compressor signals denoted as N2-1 and N2-2.

Channel B

Two N2 Compressor signals denoted as N2-3 and N2-4.

Either channel of the EEC can detect and initiate an overspeed event independently of the other channel, whether the channel is in control or not, providing two valid speed signals are available.

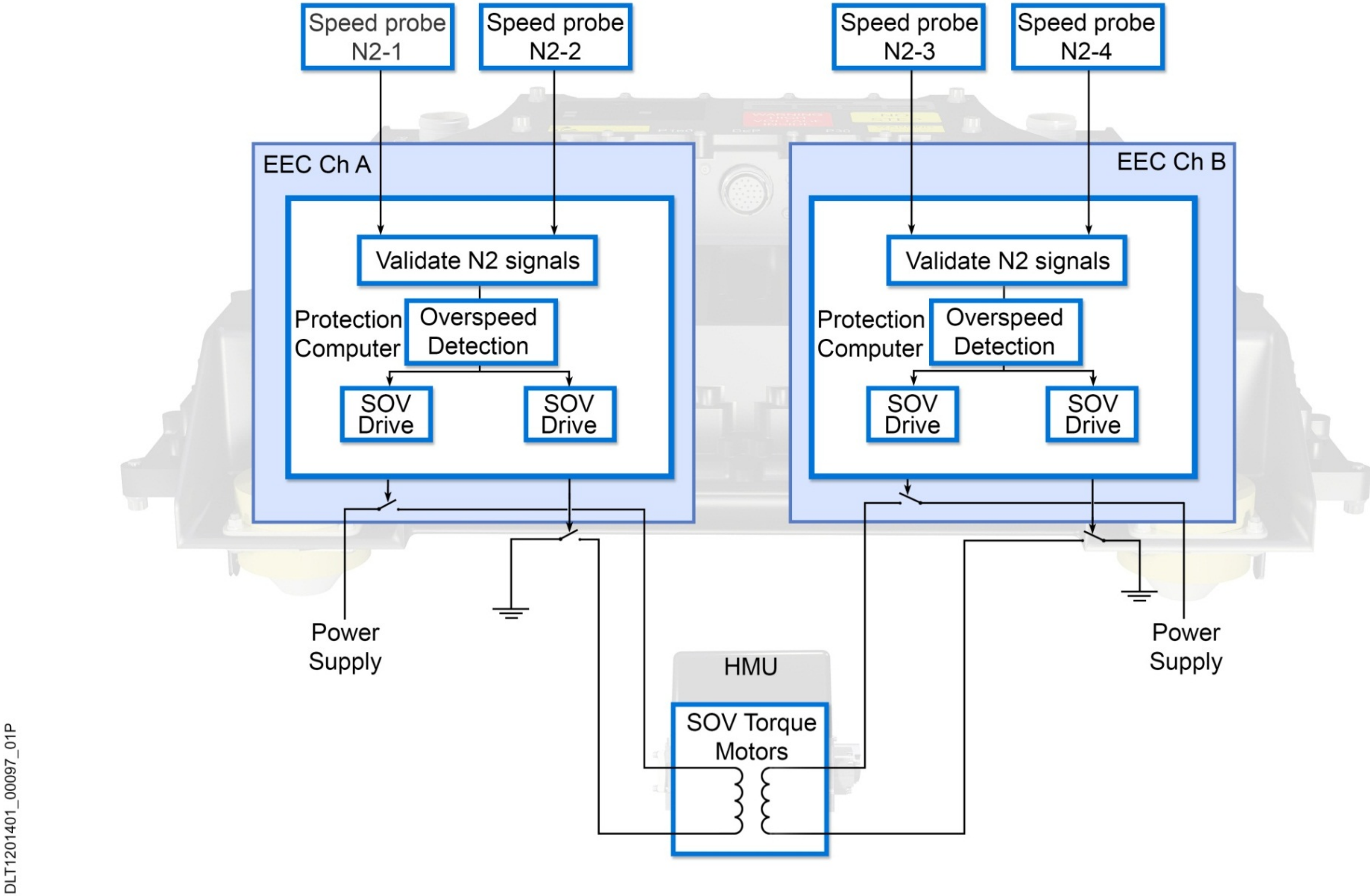
If a valid event is detected the appropriate channel of the PRSOV torque motor in the HMU is energised to stop fuel flow and initiate an engine shutdown, after which the function is latched until the EEC is reset by toggling the ON / OFF Master Lever from ON to OFF.

Signal Loss

If either of the two available speed signals is invalid then the function in that channel is disabled but the other channel will still provide protection.

BITE Checks

The IP shaft overspeed system is subject to a BITE test to confirm correct system operation.



N2 SHAFT OVERSPEED

Engine Over-Threat Protection Systems

Thrust Control Malfunction Accommodation (TCMA)

Introduction

Aircraft that have two wing mounted engines may be prone to directional control problems. If one-engine experiences an un-controlled or un-commanded high thrust when low thrust has been commanded. To protect against this the EEC provides a function that compares actual engine behaviour with that commanded and takes action if a Thrust Control Malfunction is evident.

Location

The TCM function is implemented and active in both EEC channels.

Purpose

The purpose of the TCM function is to provide protection from un-commanded thrust levels by allowing either fuel shut-off on the ground only or fuel reduction via the Pullback valve in the HMU, depending on the flight envelope, which is provided by an aircraft discrete signal from the Primary Flight Computer (PRIM).

Description

The protection computer receives the following signals which it uses to calculate either a fuel shut-off or reduction:

Engine Data

- N1 speed
- Air data P0,P20, T20
- Maximum rated take-off thrust (data entry Plug)

PRIM permission discrete

- Reduced take-off thrust inputs from the airframe: Flexible temperature, De-rate Take-off, or Climb levels
- Throttle Resolver Angle (N1)
- Commanded air bleed
- Alpha floor protection from airframe
- Weight on wheels
- Flight phase

There are three types of TCM event accommodation:

Engine shutdown (on ground)

- Engine shut down is only permitted whilst the aircraft is on the ground and take-off is not in progress.

Engine pullback sufficient to prevent TCM threat (in air)

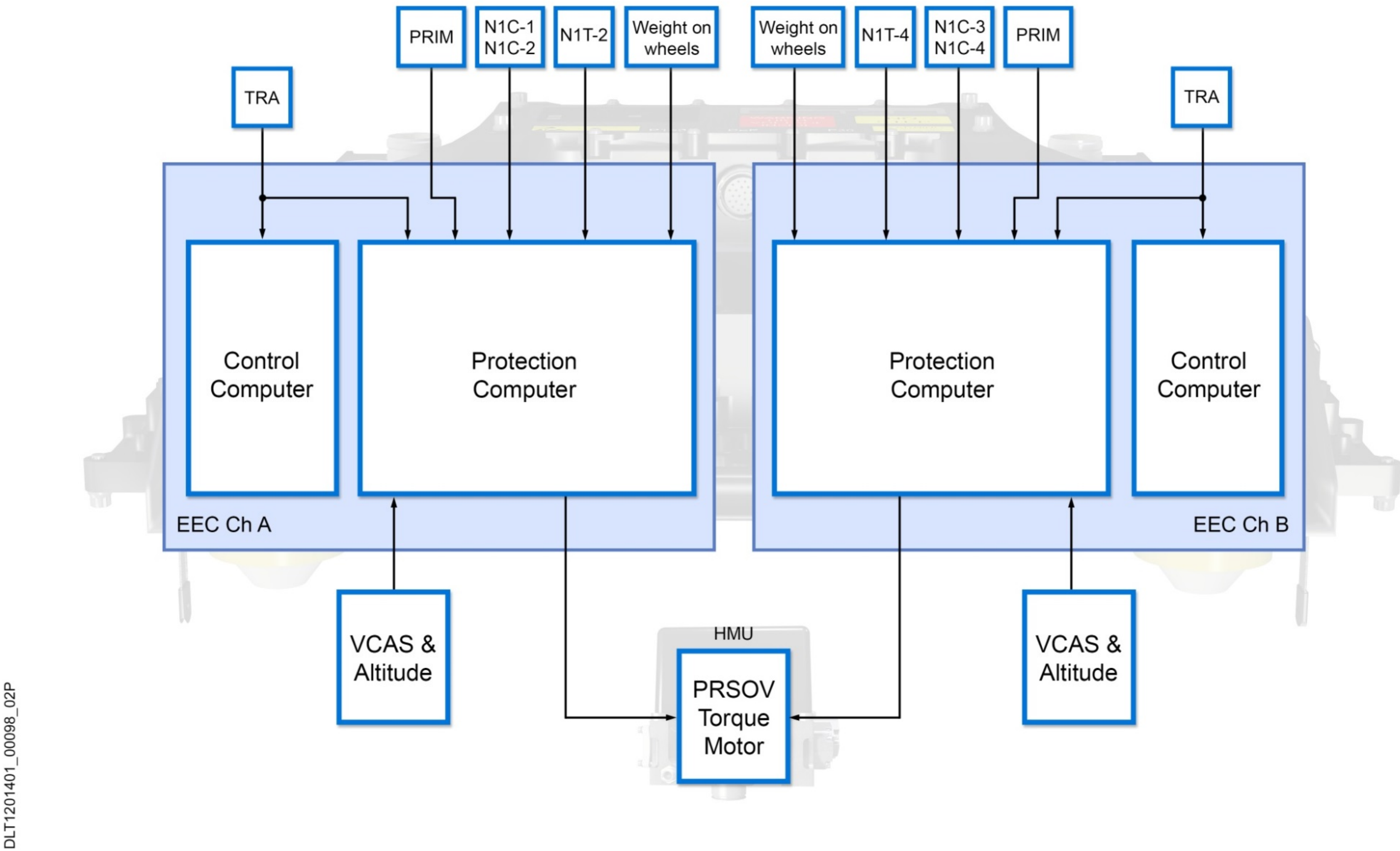
- Engine pullback is permitted whilst the aircraft is in approach mode to landing. The pullback command causes the EEC to modulate a pullback servo valve which reduces fuel flow to the engine, sufficient to remove the TCM threat.

Engine cap sufficient to prevent shutdown from over speed (on ground or in air)

- Engine cap, prevents the engine from exceeding the over speed shut down threshold by capping the N1 marginally below the redline speed.

BITE check

The TCM system is subject to a bite check at engine start up to confirm correct operation.



THRUST CONTROL MALFUNCTION (TCM)

Turbine Cavities Overheat Detection System

Location

TCAF1 and TCAF2 use single unit dual thermocouples in two positions. The TCAF1 and TCAF 2 thermocouples are located 25° left and right from TDC looking from the rear on the HP / IP turbine case, TCAF2 thermocouple being 40% longer. The single TCAR duplex thermocouple is located 6° right from TDC looking from the rear on the IP turbine case, located inside one of the LP NGVs. A further temperature sensor 'TVent' has been proposed, and is to be located in zone 3 on the combined HP / IP bearing chamber vent pipe, (no detailed information was available when writing these notes). All of the thermocouples connect via a duplex sensor with two channels of the EEC.

Purpose

The System is designed to generate alert and maintenance messages to the aircraft cockpit when the operating air temperatures in the turbine cavities exceed nominal operational values.

Description

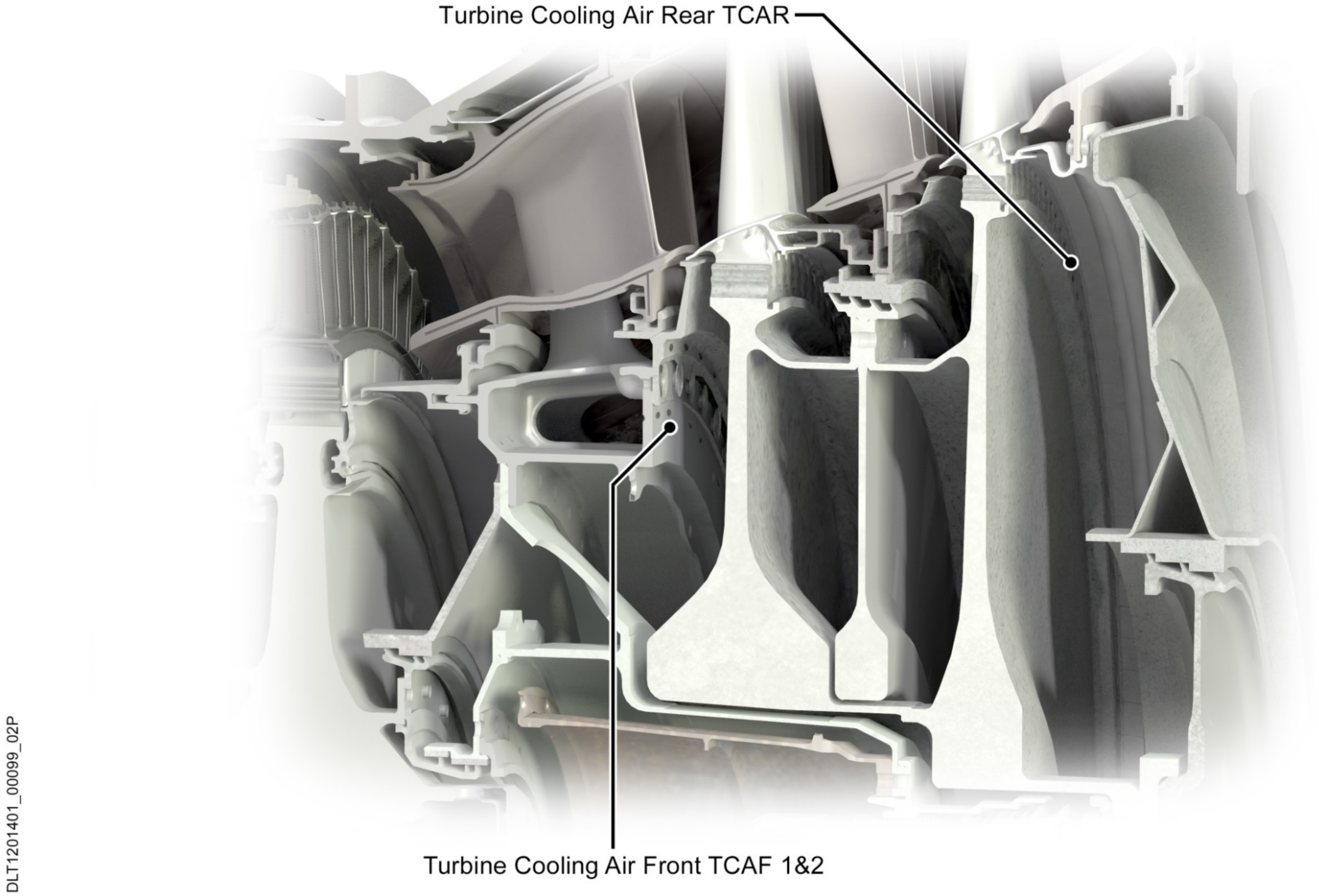
The TCAF1 and TCAR Thermocouples monitor the airflow temperatures around the HP and IP turbine area. The thermocouple signals are sent to both channels of the EEC by electrical harnesses. These signals are continually compared to limits in the EEC software. If the turbine temperatures exceed the limits for more than a given time period the EEC will send the overheat signal through the aircraft systems to provide the flight crew with an alert message (Turbine Overheat Message).

TCAF2 thermocouples monitor the airflow temperatures around the HP and IP turbine area, and are designed to detect an oil fire caused by fractured HP / IP oil pipe or leakage from the rear of the HP / IP bearing chamber.

The system is not normally capable of automatically retarding or shutting-down the engine. The pilot is required to respond within 5 minutes to pull back the engine, for which a warning has been issued. The warning messages and indications for TCAF1, TCAF2 and TCAR are the same (ENG x TURBINE OVHT)

N2 Shaft Breakage Detection

If HP / IP turbine area overheat enable signal logic is detected by the EEC, any subsequent unscheduled decrease in IP compressor speed will result in an engine shut down.



TURBINE CAVITIES OVERHEAT DETECTION SYSTEM

Engine Health Monitoring System

Introduction

The Trent XWB Engine Monitoring System (EMS) is part of the EECS. The EMS has no controlling function but rather, employs a suite of sensors to provide data that permits the aircraft and associated Ground Support Stations (GSS) to predict, diagnose and assess impending engine problems before a failure occurs. The purpose of the health monitoring system is to ensure that Rolls-Royce and the airlines have the capability to predict maintenance requirements to maximise the operational benefits. The monitoring system will provide a flexible means of capturing engine data to allow Rolls-Royce to investigate unplanned events across the whole engine fleet.

Description

The EMS consists of a series of sensors that measure temperature, pressure, speed and vibration throughout the engine. The EMS sensors input data to a fancase mounted Engine Monitoring Unit (EMU), where it is processed and made available to the aircraft and / or GSS.

The EMU is made up of a Power Supply Module (PSM), two Signal Process Monitors (SPM1 and 2) together with two Main Processing Modules (MPM1 and 2).

The EMU will support the following functionality:

- Vibration levels of N1, N2 and N3 which are displayed on the ENG page of the lower ECAM screen.
- LP Trim Balance; the EMU supports interactive maintenance operation to allow LP shaft trim balance operation's to be performed.
- Engine Trending and Event Capture System (for report generation and transmission).
- Module hours / cycle counting.

Engine Trending and Event Capture System

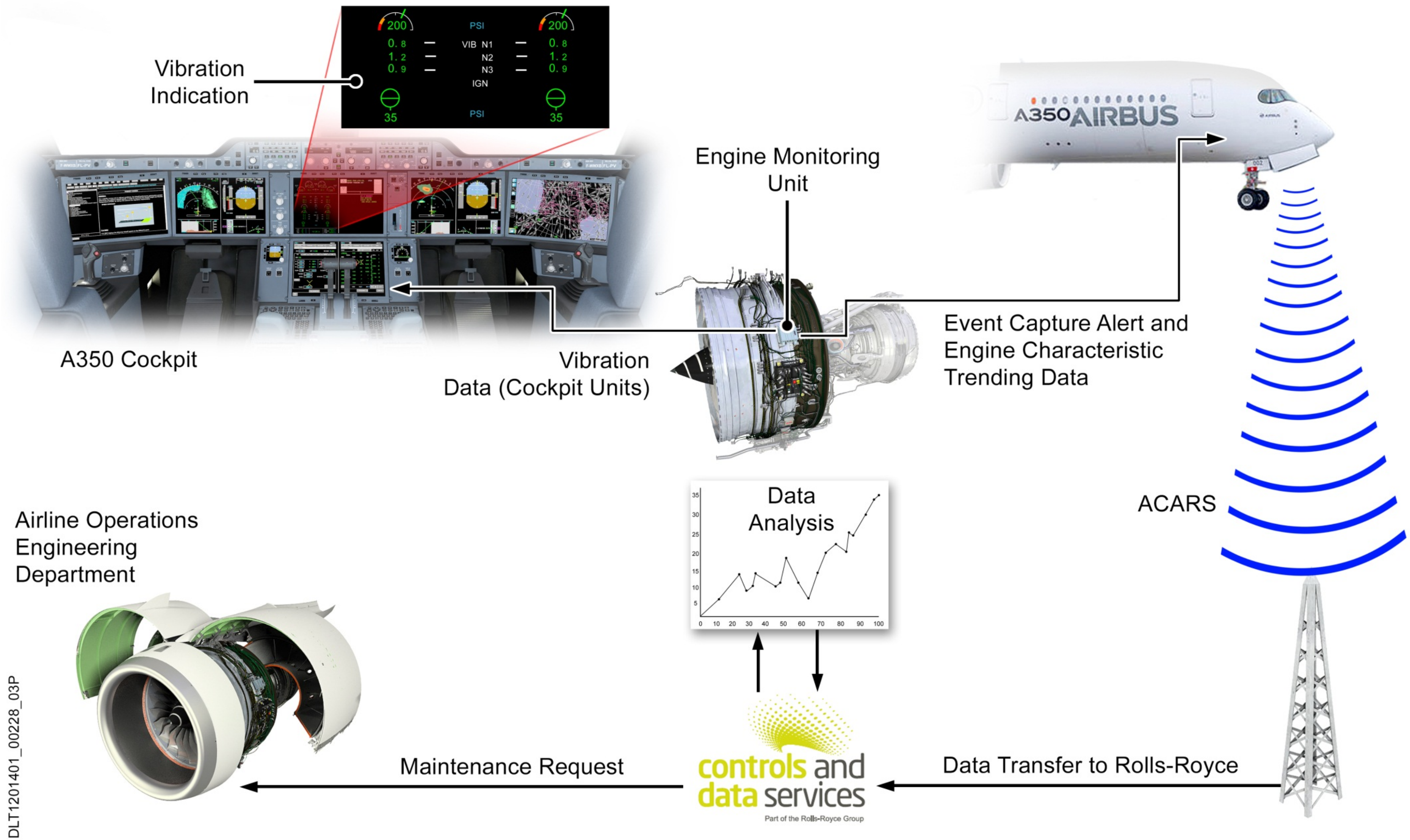
Engine Trending and Event Capture System (ETrECS) is responsible for the generation of the Engine Characteristic Trending (ECT) report and capability to detect engine problematic events. Event Capture (EC) in this instance is not history dependent, and simply relies on a set of relationships being met at that time. EC alerts will be sent back during flight via Aircraft Communications Addressing and Reporting System (ACARS).

The ECT report is a flexible tool to facilitate the capture, storage and transmission of data supporting a wide range of engine health management techniques. The full ECT report will be sent back on landing following engine shutdown.

Function

The ECT report and EC alerting should function such that any available parameter individually monitored or grouped together within the EMU could be analysed by Rolls-Royce, Controls & Data Services (CDS) or the airline.

This may result in maintenance requests received by the airline, to rectify a fault or investigate the possible cause of any deviation of a readout or performance change.



ENGINE MONITORING SYSTEM

Engine Monitoring Unit (EMU)

Introduction

The Engine Monitoring Unit (EMU) forms part of the aircraft Health Monitoring System. It collects engine data before passing it to the OMS.

Location

The EMU is located on the left side of the LP Compressor Case.

Purpose

The EMU has two (2) purposes. The first is to provide the cockpit with an indication of engine vibration levels and the second to monitor the engine health.

Description

The EMU is supplied by 115V AC power from the aircraft main AC bus. The EMU is a single channel device and the data needed to support the EMU functions, are transmitted to the aircraft AFDX Network via a bi-directional ARINC 664 Digital Data Bus.

A charge amplifier inside the EMU receives hard-wired broadband analogue vibration data from the following vibration transducer.

- A dual channel vibration transducer mounted on the engine intermediate case (N1, N2 & N3 compressor vibration).

Together with the hardwired speed signals from the EEC the EMU calculates the vibration levels and passes them to the aircraft via the AFDX.

Monitoring of the LP Compressor vibration also enables the EMU to trend the vibration and provide Trim Balance data to the aircraft and thus the airline and Rolls-Royce for forward maintenance planning.

Inputs and Outputs:

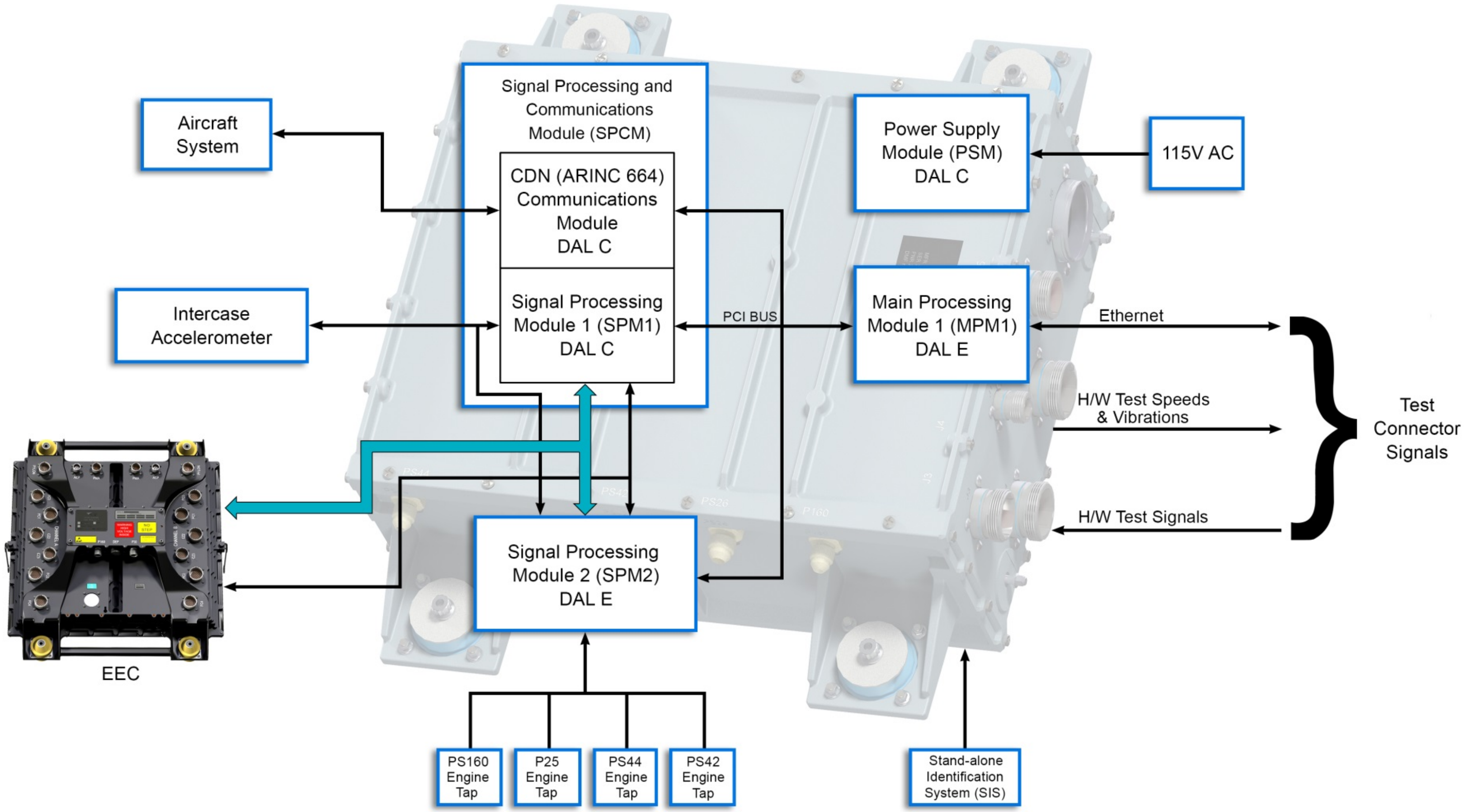
The following pressures are input directly to the EMU:

- Fan Exit Pressure, PS160 – Fan damage detection.
- IPC Exit Pressure, PS25 – Core damage detection.
- IPT Static Pressure, PS42 – Performance.
- LPT Static Pressure, PS44 – Performance.

The EMU uses five analogue speed signals, which are provided and hardwired from the EEC. These are:

- Two N1C speed signals.
- Two N2 speed signals.
- One N3 speed signal.

In addition, Channel B of the EEC transmits additional digital data to the EMU via a two-way RS422 digital bus.



ENGINE MONITORING UNIT

Interactive Maintenance

Interactive Maintenance is provided by the aircraft and EEC interface to assist maintenance personnel to confirm, in a safe environment, the presence of faults and to verify that maintenance actions have returned the system to correct operation.

In addition, the EEC assists by semi-automating certain maintenance procedures and allows data held within the EEC to be reported and where necessary reset.

The EEC provides the following Interactive functions and reports any faults detected to the Central Maintenance System (CMS): -

System Test. The EEC invokes its normal power up checks followed by electrical checks of solenoid and torque motor drives (with the exception of the Starter Air Valve (SAV drive). A period of passive monitoring to detect failures is performed for the remainder of the test duration.

Audible Test of the Igniters. The EEC energies' the selected igniter in order to allow the operator to audibly confirm its operation.

Variable Stator Vanes Actuator Test. The EEC instructs the operator to crank the engine. Once cranking the EEC schedules the VSVs to different positions to determine if they are responding correctly.

Fuel Metering Valve Test. The EEC instructs the operator to crank the engine. Once cranking the EEC commands the FMV to different positions to determine if it is responding correctly.

Bleed Valve Test. The EEC instructs the operator to start the engine. Once running the EEC commands each bleed valve to operate in sequence to determine if they are responding correctly.

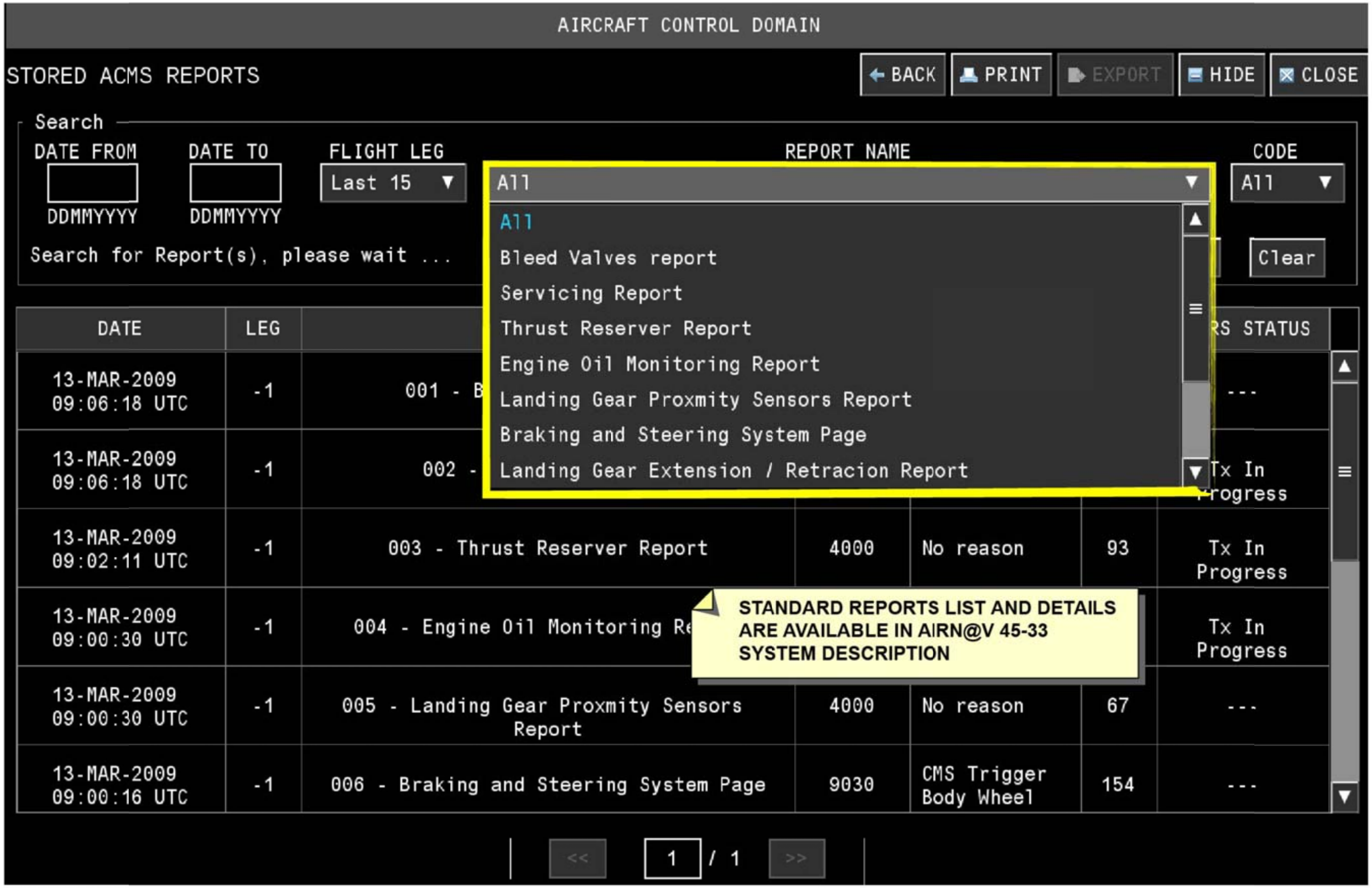
Harness Test. The EEC reduces the fault confirmation time for its Failure Messages associated with engine harnesses allowing failures to be rapidly confirmed.

Engine Core Washing Procedure. The EEC instructs the operator to crank the engine. Once cranking, the EEC moves the VSVs to their open position to allow optimum washing of the engine core. The EEC displays the remaining time allowed for washing the core before instructing the crank to be stopped.

EEC Configuration Report. Engine configuration data resides in the Data Entry Plug and is copied into EEC Non-Volatile Memory (NVM) after ground power up of the EEC.

Shaft Speed Exceedance Report. The EEC reports the shaft speed exceedance data recorded during the last engine run to the CMS. For the shaft speeds (NL, NI and NH) if the redline was exceeded on the last engine run, the report displays the peak values reached and the total length of time above the redline limit. The operator is given the option to reset the values. The values can either be reset via this Interactive function or by performing a ground engine start.

EGT Exceedance Report. If the EGT redline was exceeded on the last engine run, the report displays the peak value reached and the total length of time above the redline limit. The operator is given the option to reset the values. The values can either be reset via this Interactive function or by performing a ground engine start.



EEC MAINTENANCE DISPLAY

EEC Software Loading

Purpose

EEC test ports are provided to allow test equipment to interface to the EEC in order to monitor and change software. The test ports also provide an alternative means to download new software programmes to the EEC aircraft use when the on board data load system is not available.

A reprogramming baulking feature has been included in the design of this unit that ensures that only software designated as suitable for this unit is allowed to be programmed. The baulking feature shall ensure that Trent 1000 software cannot be inadvertently loaded into this unit and that Trent XWB software suitable for this unit cannot be programmed into an existing Trent 1000 unit.

Description

The software is loaded into the EEC via the test port input of the EEC and a “bootstrap” programme embedded within the EEC provides the capability to load software. Individual software builds are loaded into each of the three processors (control processor, safety processor and communication processor). A configuration identity is created for the combination of all these software builds. The format of this identifier is defined and agreed with the aircraft manufacturer to be the correct format used for all software systems on the aircraft.

The combined build standard of the EEC software has a unique part number identification of the format:

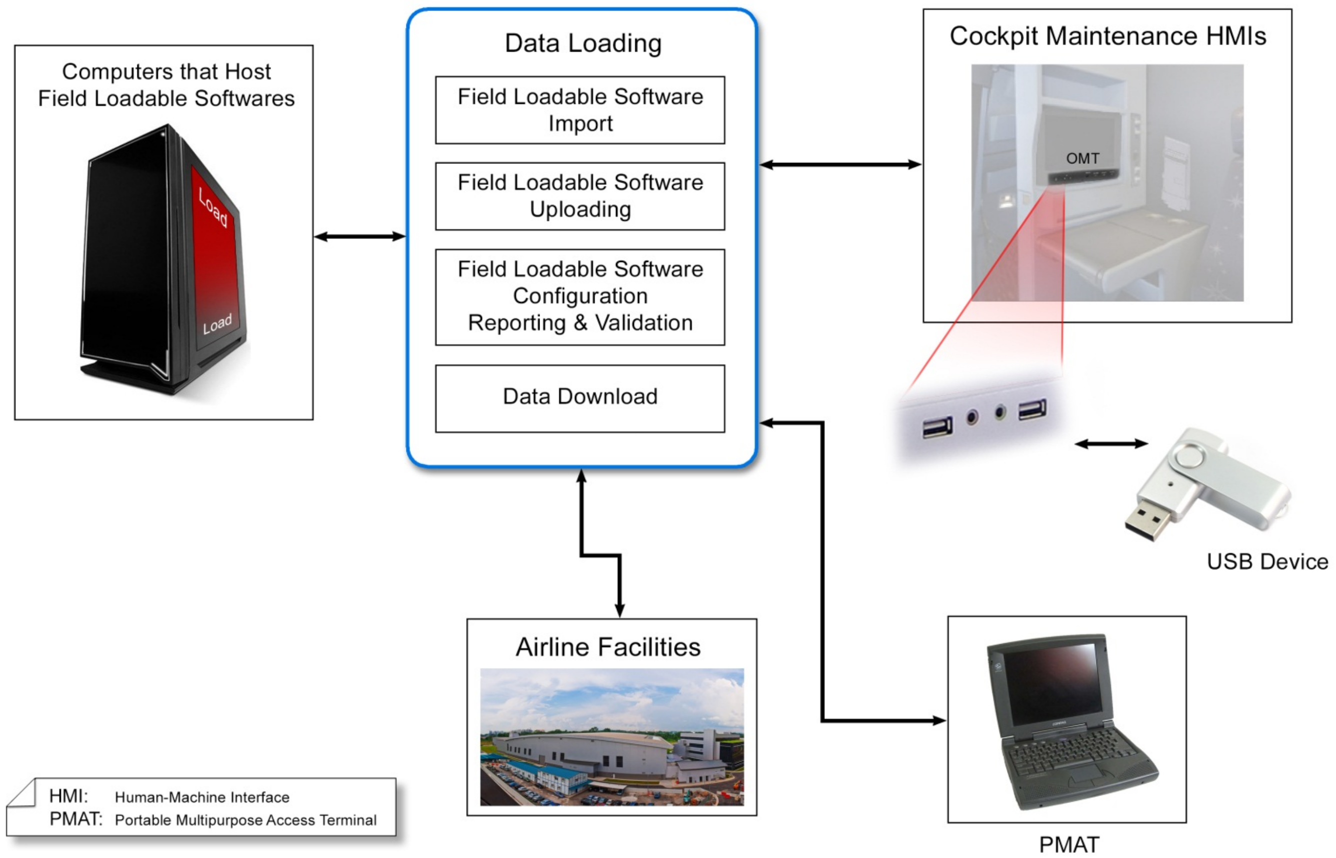
RRYccXWBxxxxxxx

Where:

- RRY = invariant identifier for Rolls-Royce.
- “cc” = Checksum digits calculated for loadable software build.
- XWB = invariant identifier for XWB software build.
- “xxxxxxx” = unique numerical identifier for a specific software build.

The software build standard part number will be programmed into the EEC as part of the software download process to allow it to be communicated externally.

A Stand Alone Identification System (SIS) allows the software part number of the EEC to be interrogated when the unit is on the engine.



DLT1201401_00215_02P

EEC SOFTWARE LOADING

Data Entry Plug (DEP)

Location

The Data Entry Plug (DEP) is semi-permanently fastened to the engine fan case by a lanyard and plugged into the bottom of the EEC by a dedicated connector.

Purpose

The DEP contains data that is specific to the engine it is fitted to. To control all possible engine configurations regardless of individual engine characteristics the EEC reads the information stored in the DEP.

Description

The DEP contains two identical Electrically Erasable Programmable Read Only Memory (EEPROM) devices, one for each channel of the EEC.

During 'power up' the EEC reads and validates the content of the DEP, storing the information in its internal Non-Volatile Memory (NVM). If during subsequent 'power ups' the validated information differs from the data already stored in the NVM of the EEC, the NVM is overwritten.

DEP Contents

Both DEP EEPROMs are programmed with identical data

- **Engine Serial Number (ESN)**

  The ESN is a five-digit number with a valid range between 20000 and 29999. It used to indicate to the EEC which engine it is mounted on.

- **Engine Ratings**

  This is a numerical value between 1 and 61 that indicates to the EEC which of the rating and bump tables stored within the EEC is selected for use.

- **Turbine Gas Temperature (TGT) Trim**

  The EEC uses the TGT value to calculate the corrected % Thrust value (THR). The THR is used for engine control as the THR has a direct relation with the EGT. The EEC sends signals to the EMU and to the cockpit where the ECAM shows an EGT indication as a value of degrees Celsius. The EEC uses data held by the Data Entry Plug (DEP) to adjust (trim) the THR signal. This makes all engines operate to the same EGT limit indication shown, as shown on the cockpit EGT display. The quantity of trim that is necessary is programmed into the DEP after engine test. Trimming THR in this way makes every engine of a given standard appear identical to the airframe.

- **Engine Standard**

  The DEP gives the EEC an indication of the engine standard.

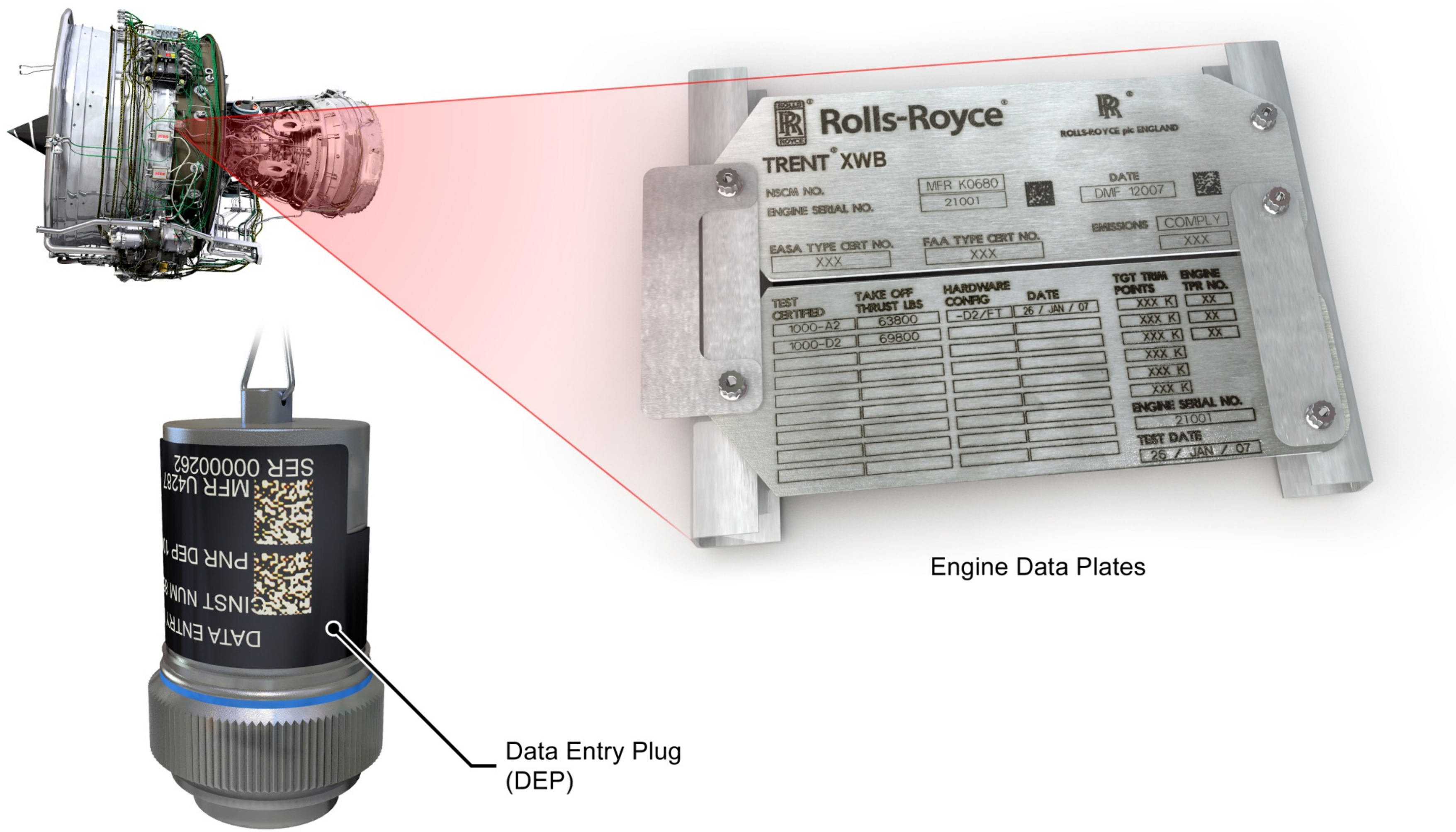
- **Intermix (provision)**

  Intermix data tells the EEC if different engine standards are / are not installed on the same airframe.

DEP Programming

The DEP can be (re) programmed with valid data using a dedicated DEP programmer. The DEP should only be reprogrammed if:

- The DEP is new to the engine.
- There has been a change of engine rating.
- The trim values have changed post overhaul.



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ENGINE DATA PLATE & DEP

EEC Maintenance

DEP Programmer

AMP TASK TRENTXWB-A-73-21-12-00A01-752A-A

AMP TASK TRENTXWB-A-73-21-12-00A01-752A-D

Introduction

The Trent XWB Data Entry Plug (DEP) Programmer (UT2011) is a portable, self-contained unit used for programming and analyzing of Trent XWB Data Entry Plugs. All components necessary for operation have been housed in a rugged transport case (UT1975) with the capability of producing a hard copy via an integral printer.

The Programmer comprises of two main components referred to as the “Handheld Unit” (UT1972) and the “Trent XWB Data Entry Plug Programmer Interface” (UT2012). These components are interconnected using a cable (UT1974). The unit is supplied in a storage case with a battery charging cradle (UT1973), an AC mains adapter (UT1593), two spare paper rolls and a User Manual (EIR19544).

Battery

The Handheld Unit is powered from a Lithium Ion rechargeable battery pack to provide up to 10 hours continuous use (the battery should be in its disconnected stowage position during prolonged storage periods). The unit should be fully charged before use (this may take up to 8 hours), alternatively the unit may be powered externally using the charging cradle and mains adapter.

Connecting the Interface Cable

Connect (or remove) the Interface Cable (UT1974) by locating the small interface cable connector with the red dot facing the top and aligned centrally between the connector and socket.

Applying Power

If the display on the Handheld Unit is not visible, press and release one of the Soft Trigger Buttons on either side of the unit and the Start Screen will appear on the display within a few seconds. If the Handheld Unit is left ON but is not actively communicating with the Trent XWB DEP Programmer Interface, it will power down automatically after approximately five minutes.

Software

DEP programmer software will be supplied on a smartcard as the programming specification changes. To install the new software, complete the following:

- Place the unit into a suspended condition by pressing the Suspend button on the Start Screen.
- Ensure the Interface Cable is not connected.
- Remove the Bottom End Cap by using a flat blade screwdriver to rotate the fasteners ¼ turn.
- Press the Eject Button (just above the Reset switch) to remove the smart card from the memory slot.
- Install the new smart card making sure that the orientation is correct.
- Carefully press and release the Reset Switch, using a small screwdriver after twenty seconds the Start Screen will again be displayed.
- View software page to ensure the new software version matches that on the new software card.

DEP PROGRAMMER

EEC Maintenance

DEP Programming

Operation

The DEP should only be (re) programmed when:

- The DEP is new to the engine.
- There has been a change of engine rating.
- The trim values have changes post overhaul.

The Unit has a touch sensitive screen so when using the Trent XWB DEP Programmer, the operator will be required to touch, press gently or tap active areas of the screen, using a finger or the dedicated touch screen stylus supplied.

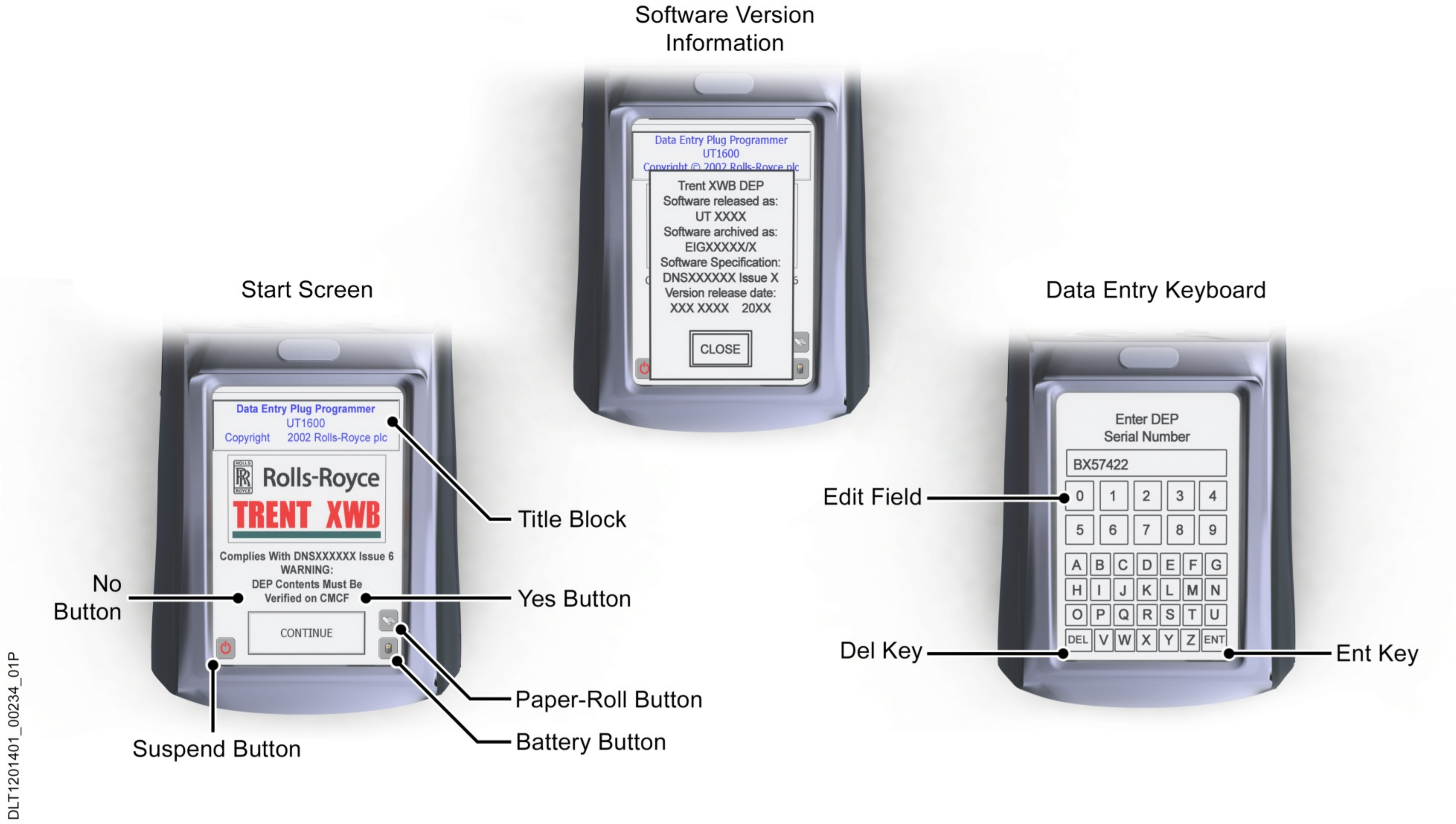
Before using the Trent XWB DEP Programmer the operator is advised to check that the Programmer is fitted with the correct version of software. The software version is visible on any printout produced by the Handheld Unit and is also displayed by tapping on the Title area at the top of the start screen. Check the displayed software UT number against that required by the Aircraft Maintenance Procedures (AMP). Press the Close button after viewing of the Software version information.

Performing DEP Analysis

Note: Always refer to the AMP for further detailed information.

To perform DEP Analysis / Programming the following actions are required:

- Check that the correct software version is installed.
- Adjust the contrast and brightness of the Handheld Unit display, if required.
- Adjust the Date and Time setting of the Handheld Unit, if required.
- Complete the DEP Interface startup Self-Test.
- Connect a Trent XWB DEP.
- Enter the DEP Serial Number (using Data Entry Keypad).
- Select from options to Erase the DEP. Edit DEP Contents, and, Print the DEP Contents or Remove the DEP.



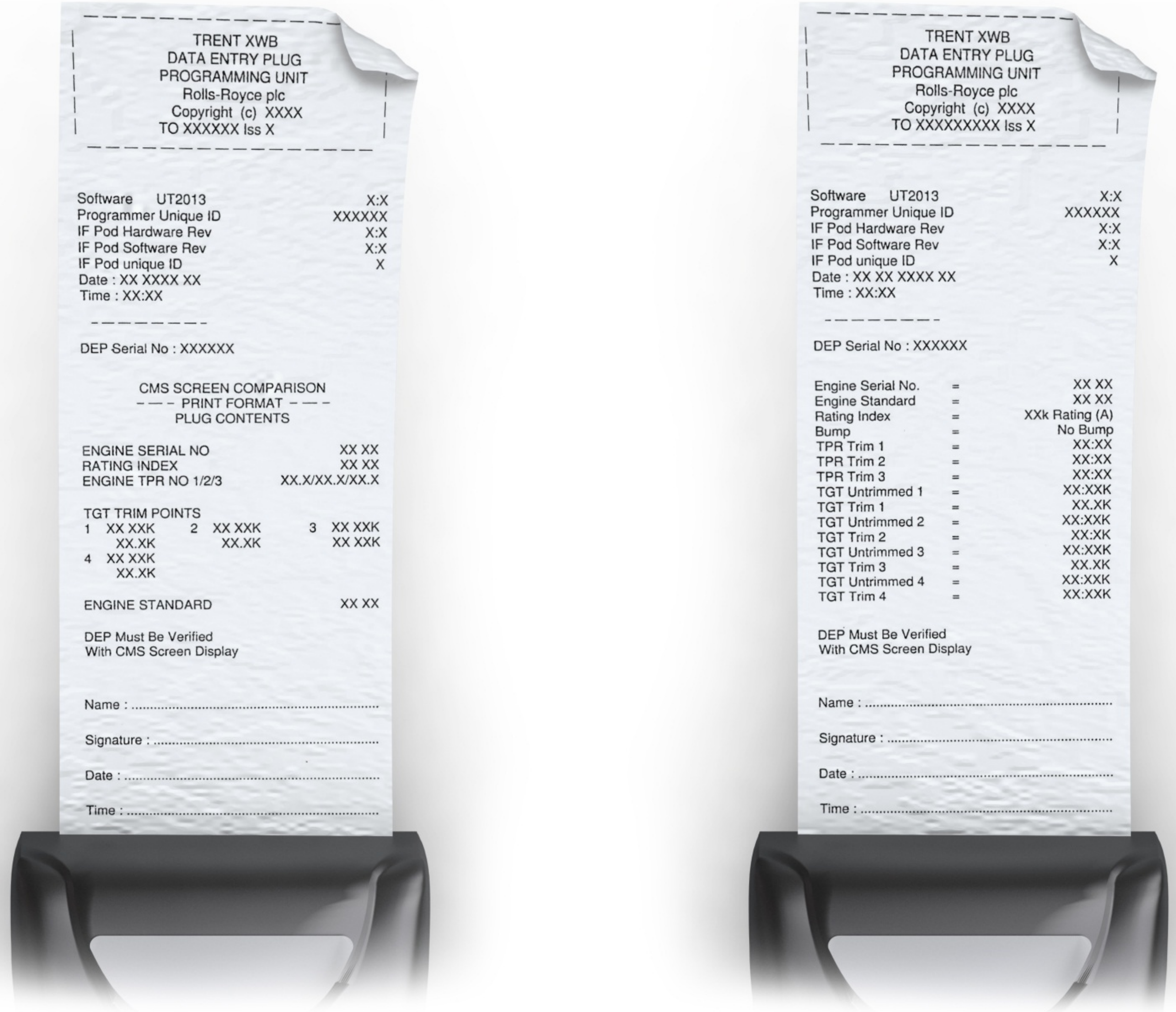
DEP PROGRAMMING

Programmer Printout

A printout of the information contained on the DEP is available as an option.

If you are required to do so as part of the AMP or company procedure this should be obtained and kept for recording purposes.

A typical example is illustrated below.



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PROGRAMMER PRINTOUT

Maintenance Practices.

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Removal Procedure

Refer to the AMP for the full description.

EEC Removal

AMP Task TRENTXWB-A-73-21-11-00A01-520A-A,

TRENTXWB-A-73-21-11-00A01-520A-D.

WARNING

YOU MUST MAKE SURE THAT YOU CAN HOLD THE WEIGHT OF THE COMPONENT BEFORE YOU REMOVE/INSTALL IT. THE COMPONENT IS HEAVY. IF IT FALLS, IT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

CAUTION

YOU MUST NOT USE PLIERS WITH METAL JAWS TO LOOSEN THE CONNECTORS. THE METAL JAWS WILL CAUSE DAMAGE TO THE CONNECTORS.

CAUTION

YOU MUST NOT BEND THE ELECTRICAL HARNESS TOO MUCH WHEN YOU DISCONNECT/CONNECT THE ELECTRICAL CONNECTORS. IF YOU DO NOT OBEY THIS INSTRUCTION, DAMAGE TO THE HARNESS CAN OCCUR. THIS CAN CAUSE ELECTRICAL CIRCUIT DEFECTS.

CAUTION

YOU MUST OBEY THE INSTRUCTIONS GIVEN IN THE STANDARD PRACTISE 70-12-01 WHEN YOU REMOVE / INSTALL OR DISCONNECT/CONNECT THE TUBE(S). IF YOU DO NOT DO THIS, DAMAGE TO THE TUBE(S) CAN OCCUR.

Be careful when you move the Engine Electronic Control (EEC) because it weighs 23.18 Kg (51.1 lbs).

Note

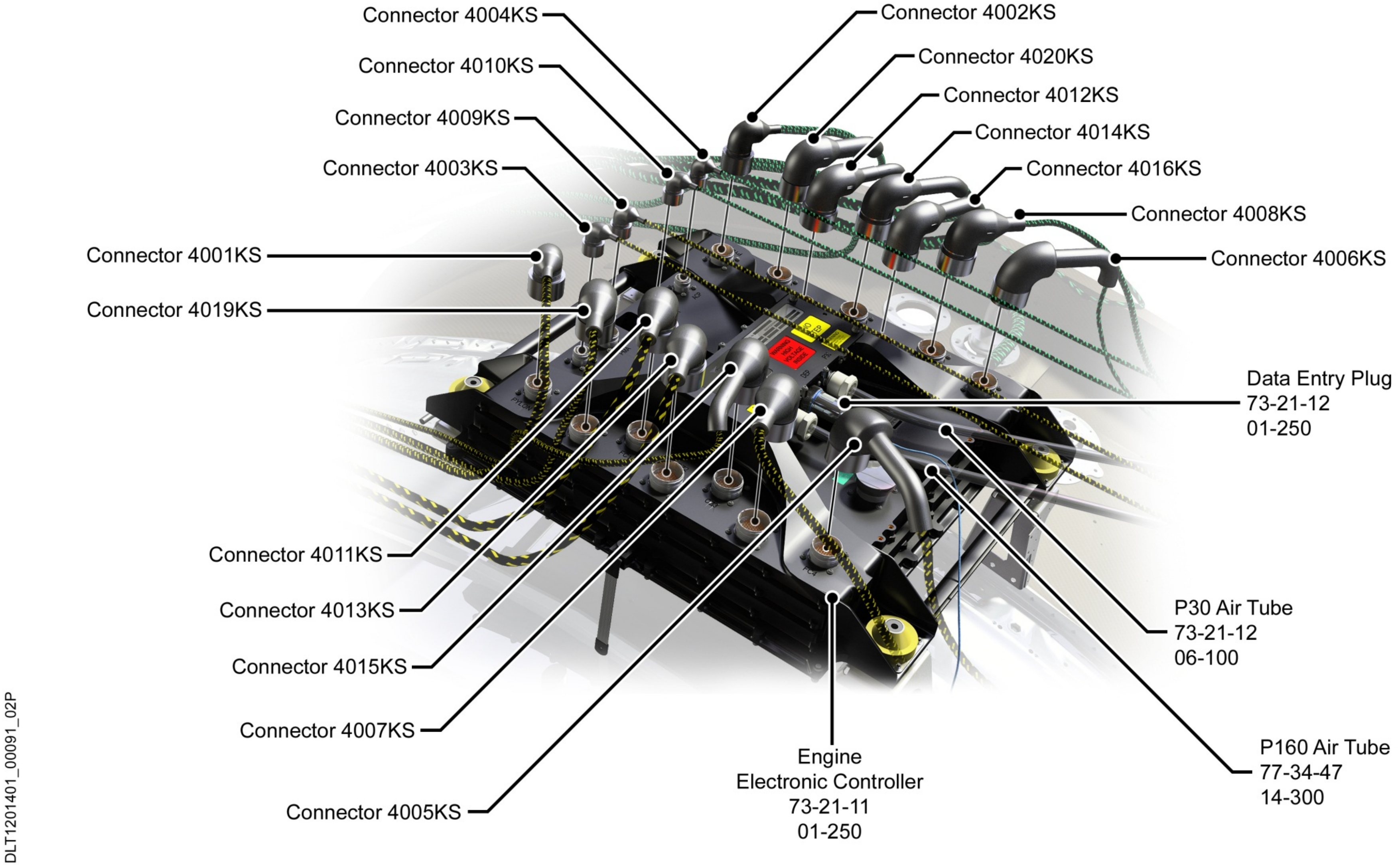
Connector 1 (part of the engine electronic control) is identified as P160 on the engine electronic control.

Note

Connector 2 (part of the engine electronic control) is identified as P30 on the engine electronic control.

MAINTENANCE TIP

The replacement of the EEC should be done as a last resort. The reseating, cleaning of the electronic plugs, and following the troubleshooting process must be adhered to at all times. This will greatly reduce the cost of EEC testing to the operator because of incorrect component replacement when troubleshooting.


Connector 4004KS
Connector 4010KS
Connector 4009KS
Connector 4003KS
Connector 4001KS
Connector 4019KS
Connector 4002KS
Connector 4020KS
Connector 4012KS
Connector 4014KS
Connector 4016KS
Connector 4008KS
Connector 4006KS
Data Entry Plug
73-21-12
01-250
P30 Air Tube
73-21-12
06-100
P160 Air Tube
77-34-47
14-300
Engine
Electronic Controller
73-21-11
01-250
Connector 4005KS
Connector 4007KS
Connector 4015KS
Connector 4013KS
Connector 4011KS
DLT1201401_00091_02P

EEC REMOVAL

Maintenance Practices.

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Removal Installation

EEC Installation

AMP Task TRENTXWB-A-73-21-11-00A01-720A-A,

TRENTXWB-A-73-21-11-00A01-720A-D.

WARNING

YOU MUST MAKE SURE THAT YOU CAN HOLD THE WEIGHT OF THE COMPONENT BEFORE YOU REMOVE/INSTALL IT. THE COMPONENT IS HEAVY. IF IT FALLS, IT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

CAUTION

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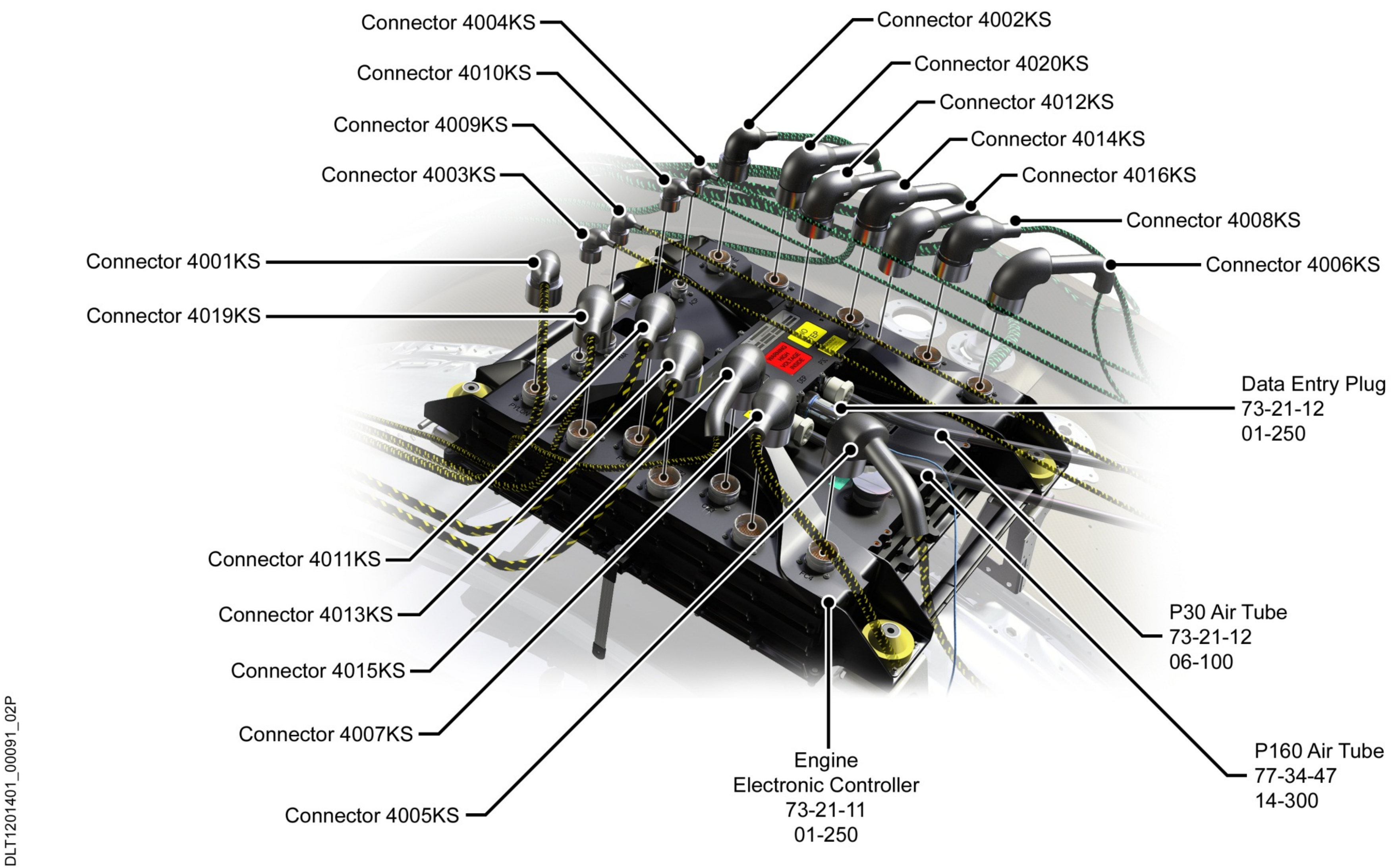


EEC INSTALLATION

On Board Maintenance – Ground or Initiated Tests

Introduction

The Initiated tests provide a means by which the maintenance personnel can conduct Engine related tests in a safe and controlled manner. The tests are called up to be carried out as and when directed by the relevant AMP or TSM procedure.

Top Level Title	Function Title
Tests	System Test
	Audible Test of the Igniters
	Variable Stator Vanes Acutator Test
	Fuel Metering Valve Test
	Bleed Valve Test
	Harness Test

System Test: Performs EEC power-up checks and electrical checks of solenoid torque motor drives followed by a short period of passive monitoring Reports all detected failures.

Harness Test: Reduces the fault confirmation time for failure message codes with engine harnesses allowing failures to be rapidly confirmed as faults at time of occurrence while the test is being performed.

DLT1201401_00238_03P

ON BOARD MAINTENANCE – GROUND OR INITIATED

Cockpit Effects

Introduction

Fault Reporting and Storage

The confirmed Failure Messages are processed into the format required by the Aircraft and transmitted to the CMS via the EECs AFDX outputs. The CMS records the Failure Messages and uses them to instigate maintenance corrective action. The EEC continuously reports the messages to the CMS until they clear. Each channel of the EEC reports the same Failure Messages, while the inter-channel communication bus is available to communicate failure messages between channels.

For engine certification, storage of Failure Messages within the EECs Non-Volatile Memory (NVM) is not performed.

Status Messages and Dispatch

The EEC determines the effect of individual and combinations of Failure Messages. The EEC allocates each combinational failure scenario to a particular dispatch category and generates an output to be transmitted to the Aircraft ECAM via the EECs AFDX outputs. The ECAM system displays Cockpit Effect engine parameters, their validity, alerts and the associated procedures.

The table below is an indication of the associated Cockpit Effect messages for this section, together with the message type (level), an explanation of its meaning, dispatch restrictions (if any) and expected actions taken by flight or ground crew.

Note: This is for your guidance only. Due to the nature of the aircraft and engine continual development these may change and Rolls-Royce cannot guarantee that they are still current and fully accurate. As such, always refer to the appropriate AMP procedure for reference.

Message	Type	Dispatch	Meaning	Crew Action
ENG Control (L/R)	Advisory	Do Not Dispatch	When on ground and the airspeed has fallen below 80 knots, the engine EEC is in a No Dispatch configuration. The EEC is operating with a limited set of engine control parameters and may not be able to control all aspects of the engine.	Do not operate
ENG Control (L/R)	Status	Do Not Dispatch	The Engine EEC is in a No Dispatch configuration. The EEC is operating with a limited set of engine conrol parameters and may not be able to control all aspects of the engine.	Do not operate
ENG EEC Mode (L/R)	Status	Short Term Dispatch Restrictions	The Engine EEC is in the alternate mode of control due to control system failure.	Repair within time scales
ENG EEC OVHT (L/R)	Status	Do Not Dispatch	The Engine EEC internal temperature exceeds the operating limit.	Do not operate
ENG EEC C1 (L/R)	Status	Short Term Dispatch Restrictions	The Engine EEC is operating with reduced redundancy.	Repair within time scales
ENG OVSPD PROT SYS (L/R)	Status	Do Not Dispatch	The Engine overspeed protection is disabled - Loss of multiple speed sensor N1, N2 or N3 roots.	Do not operate
ENG TURB OVSPD (L/R)	Status	Short Term Dispatch Restrictions	The Engine turbine overspeed detection function is disabled - Loss of all sensors at compressor or turbine end of N1 rotor.	Repair within time scales
ENG TCMA (L/R)	Status	Short Term Dispatch Restrictions	The Engine Thrust Control Malfunction Accomodation is disabled.	Repair within time scales
ENG TCMA (L/R)	Status	Unlimited Dispatch	The Engine EEC maintenance power switch (EEC POWER on the Maintenance Control Display) is currently in the TEST position.	Remove EEC power after maintenance

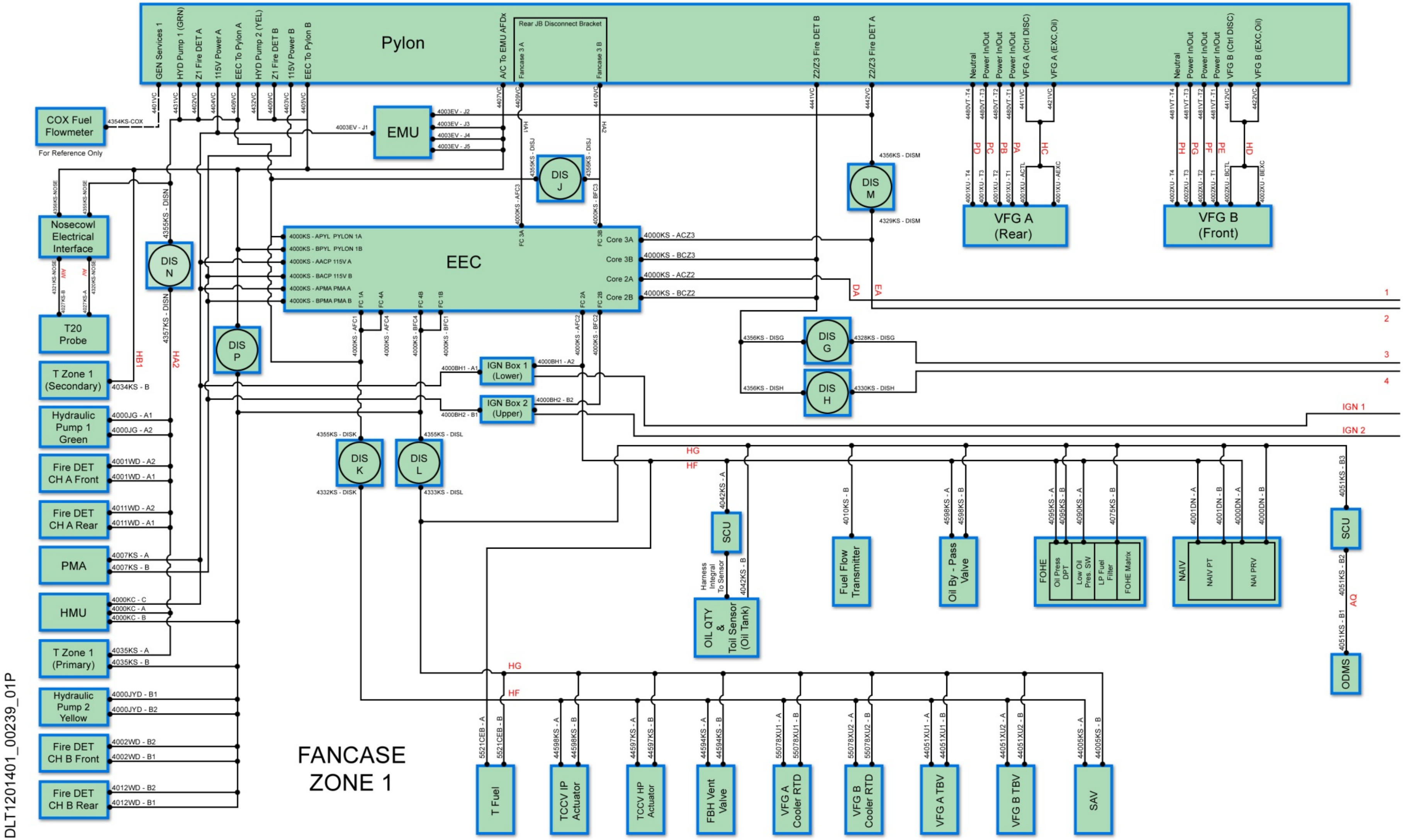
DLT1201401_00100_03P

COCKPIT MESSAGES

EEC Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.

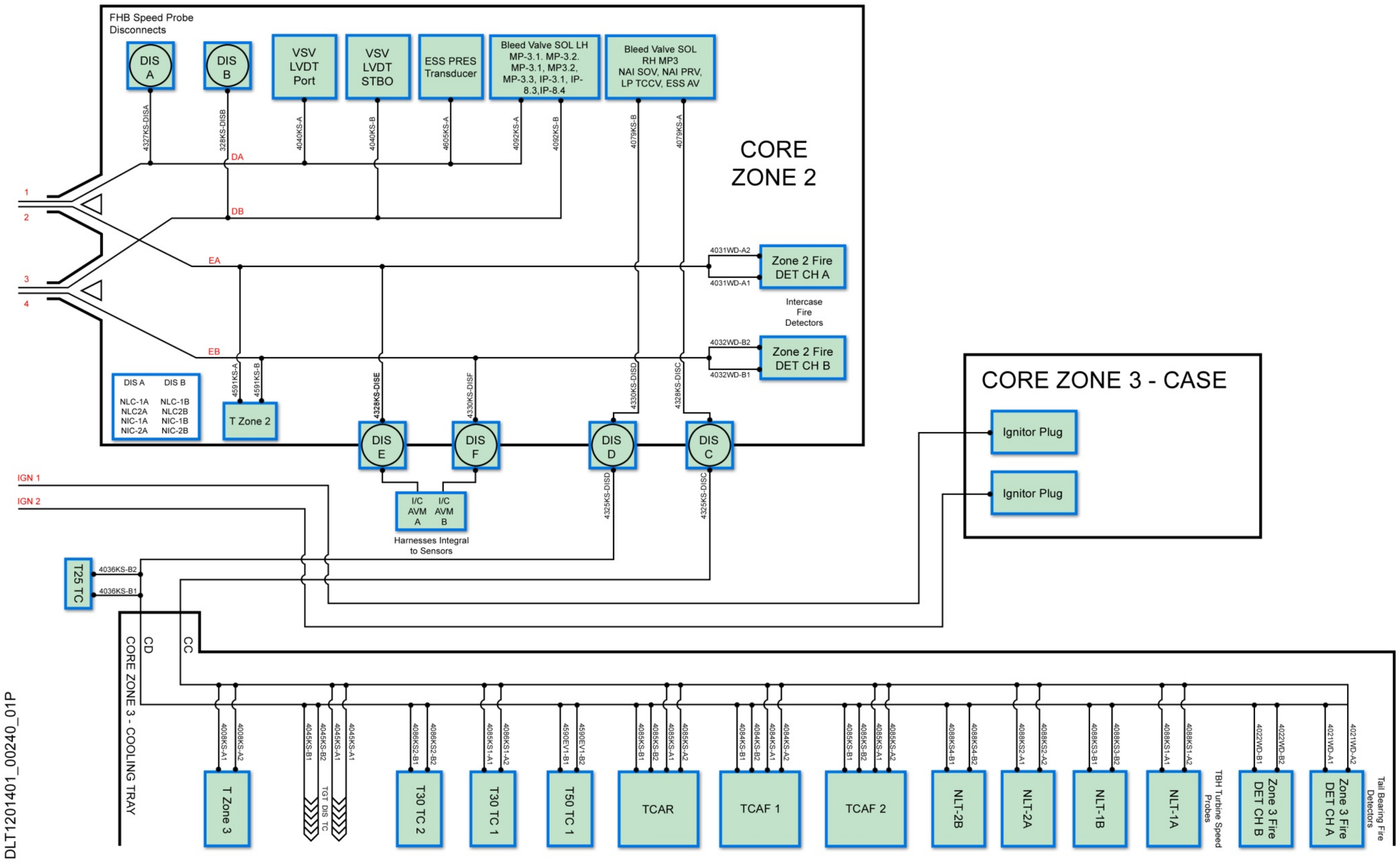


ZONE 1 WIRING DIAGRAM

EEC Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.



ZONE 2 & 3 WIRING DIAGRAM

Propulsion Control System

Objectives

At the end of this section the student will be able to:

- State the purpose of the Propulsion Control System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Propulsion Control System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Propulsion Control System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Propulsion Control System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Propulsion Control System interfaces with other engine and aircraft systems.

End of PCS Section

Section 6 - Engine Indicating System

Section 6 - Engine Indicating System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of the section the student will be able to:

- State the purpose of the engine indicating system.
- Identify the location of the engine cockpit panels and displays associated with the engine indicating system.
- Identify the location, purpose and operation of the engine sensing positions.
- Identify the location, purpose and operation of the shaft speed measurement systems.
- State the WARNINGS & CAUTIONS associated with the engine indicating system.
- Describe how the Trent XWB engine indicating system interfaces with other engine and aircraft systems.

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Cockpit Displays and Panels

Location

The engine indications are displayed on one of six Display Units (DU) in the cockpit of the A350 aircraft.

Additional panels in the cockpit allow for engine associated buttons, switches and levers to be selected.

Purpose

The purpose of the DUs is to display main engine systems flight and navigational information to the flight and maintenance crew.

The purpose of the Cockpit Panels is to provide location for aircraft flight and systems operating buttons, switches and levers.

Description

There are six full colour DUs in the cockpit, five on the forward panel and one in the centre pedestal. They are:

- Left and Right Onboard Information System (OIS) DUs.
- Captain and First Officer (F/O) Electronic Flight Instrument System (EFIS) DUs.
- Central Electronic Centralized Aircraft Monitoring (ECAM) DU.
- Pedestal mounted Multifunction Display (MFD) DU.

The DUs display information provided by the Control and Display System. This data is transmitted via the AFDX system. (Avionics Full Duplex Ethernet)

Onboard Information System (OIS)

The OIS provides the flight and maintenance crews with operational and maintenance applications and services, these are hosted in two OIS cabinets. This function allows for a fully paperless cockpit.

Electronic Flight Instrument System (EFIS)

The EFIS DUs display the main flight information for each member of the flight crew such as the Primary Flight Display (PFD) and the Navigation Display (ND). The information displayed to each crew member EFIS screen can be changed between each crew member’s screens required.

Electronic Centralized Aircraft Monitoring (ECAM)

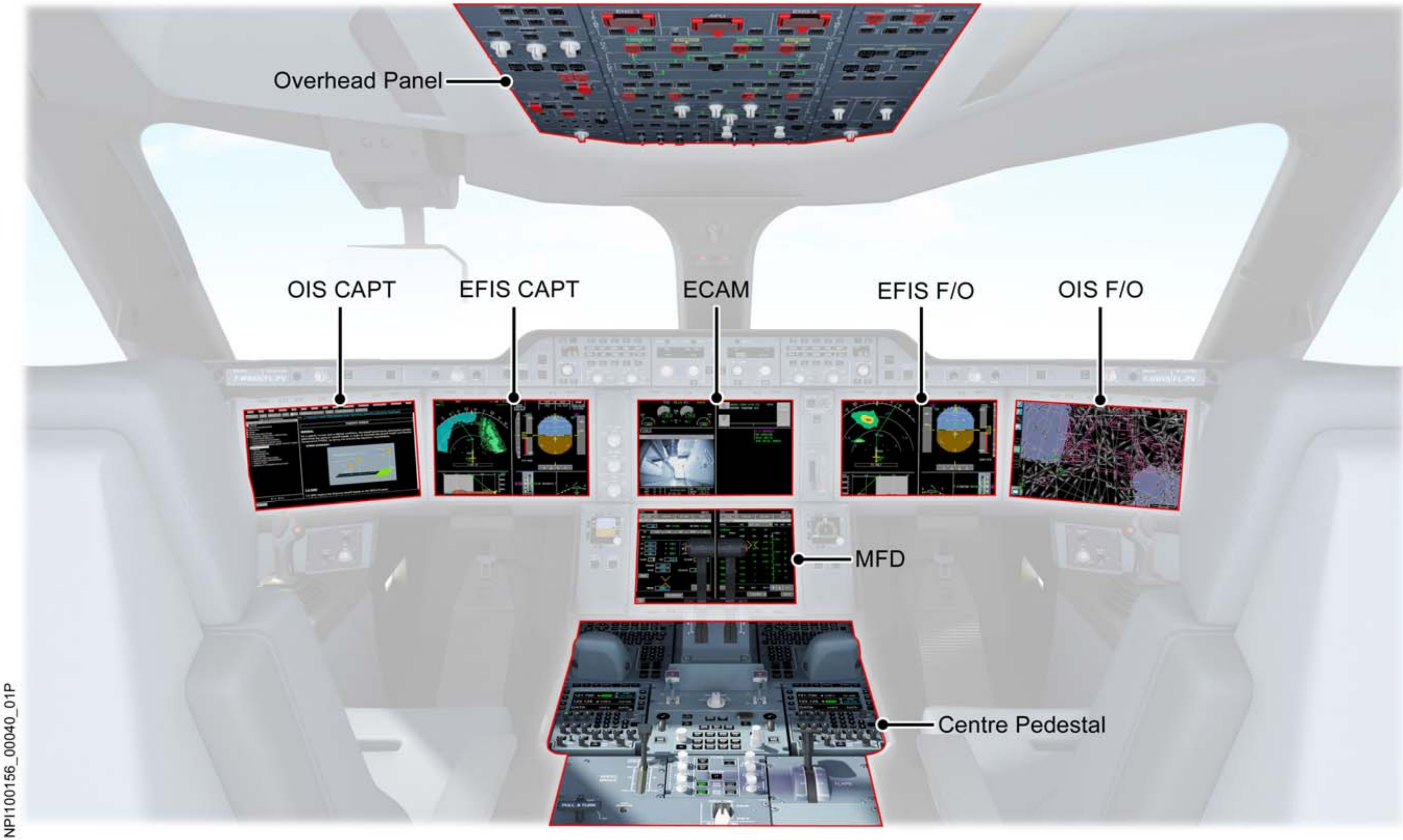
The ECAM screen displays engine parameters, Air Traffic Control (ATC) communications and warning messages.

Multifunction Display (MFD)

The MFD allows the flight crew to enter the individual flight details such as the flight plan into the aircraft system.

Head-Up Display (HUD)

The optional HUD provides guidance to the flight crew by gathering primary flight display information, and displaying engine related messages such as “Thrust Reverser Deployed” and “Auto Thrust Engaged” etc.

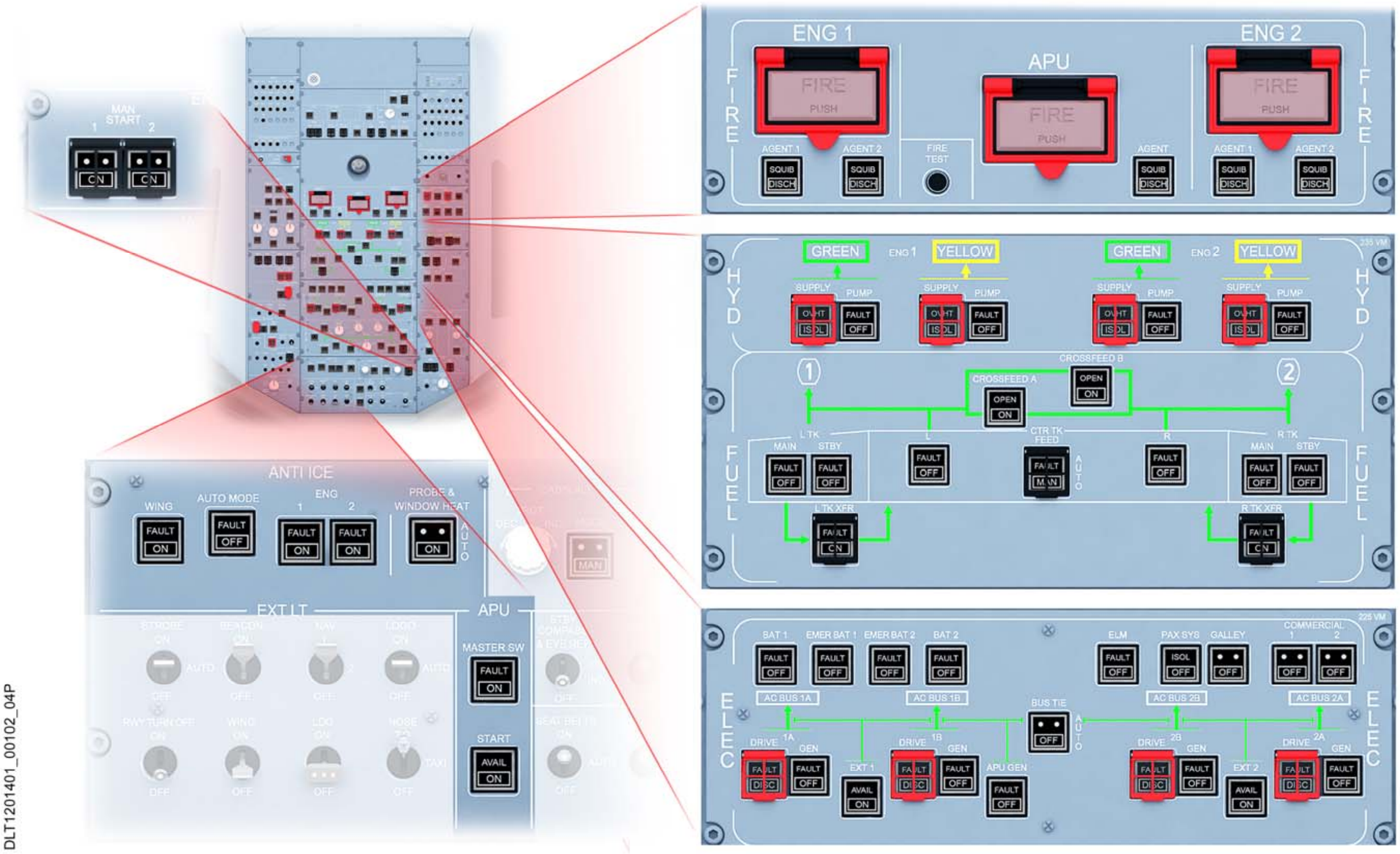


COCKPIT DISPLAYS AND PANELS

Overhead Panel

The overhead panel is located centrally and above the two flight crew members and locates operating switches and buttons for the aircraft systems such as:

- Manual engine start,
- Electrical generation,
- Auxiliary Power Unit (APU) operation,
- Fire extinguishers,
- Nacelle anti ice,
- Fuel,
- Hydraulics,
- FADEC ground power.
- Pneumatics.

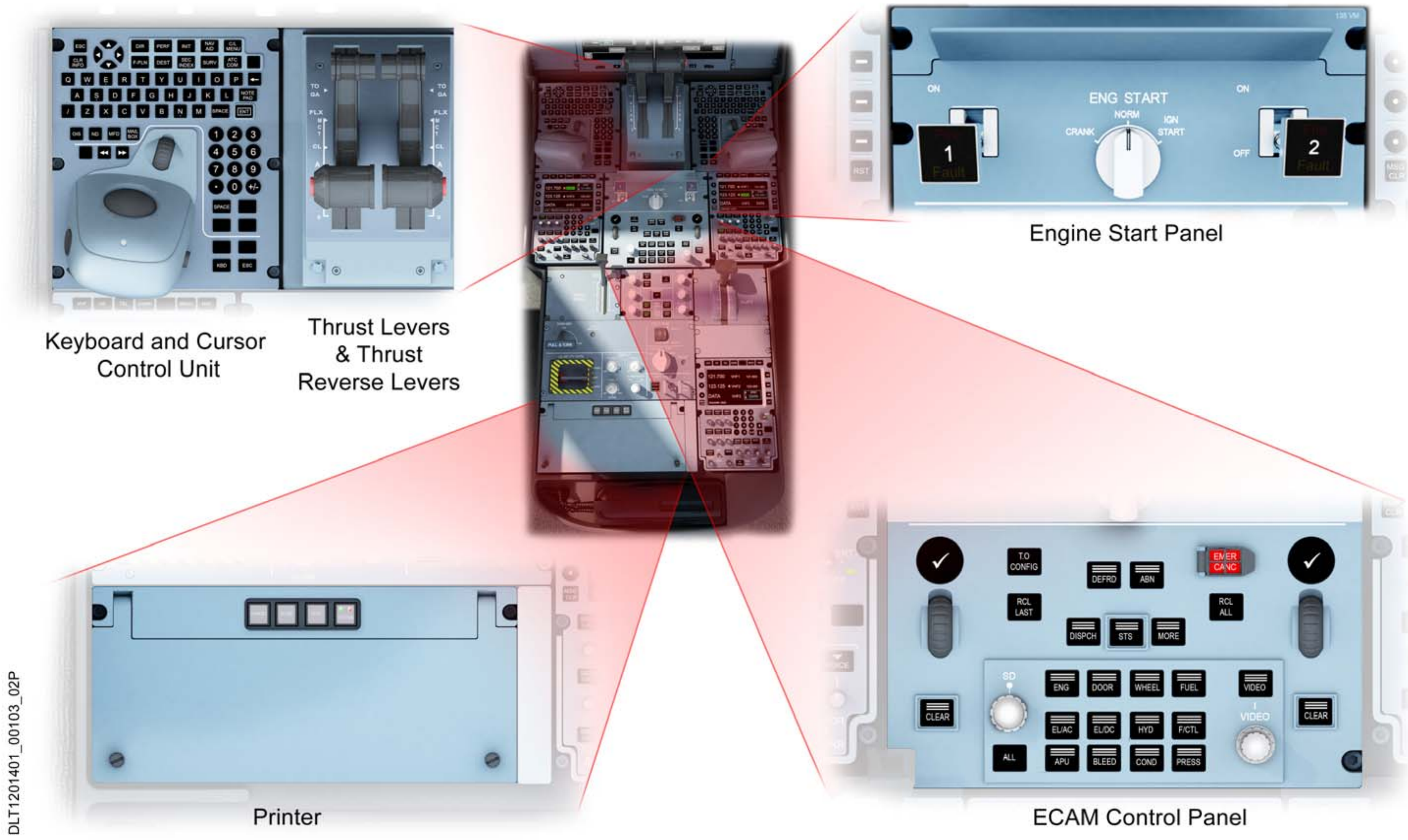


OVERHEAD PANEL

Centre Pedestal

The centre pedestal is located between the crew members and houses the following engine related operating switches and buttons for the aircraft systems such as:

- Engine start / stop Master Levers,
- Engine start mode rotary selector,
- Thrust levers,
- Thrust reverse levers,
- ECAM Control panel
- Keyboard and cursor control unit.
- Printer



CENTER PEDESTAL

ECAM Primary Engine Parameters

Location

The primary engine parameters are located on the upper left quadrant of the Electronic Centralised Aircraft Monitoring (ECAM) display on the central DU.

Purpose

The purpose is to provide a constant indication of critical engine operating parameters to the crew.

Description

The primary engine parameters are described below:

Thrust Indication (THR) – Analogue and digital indications show the crew the two thrust conditions, and thrust reverser sleeve / lock positions.

- Engine forward thrust.
- Engine reverse thrust.

Thrust is indicated as a percentage of the maximum take-off thrust from an N1 speed signal provided to the Airbus Cockpit Universal Thrust Emulator (ACUTE) by the EEC through the AFDX system.

Low Pressure System Speed (N1) – Digital indication shows the speed of the engine Low Pressure (LP) System. The LP speed is shown as a percentage of the maximum LP shaft rotational speed and is calculated by the EEC.

Exhaust Gas Temperature (EGT) – Analogue and digital indications show the corrected temperature of the turbine, which is calculated by the EEC.

Other primary parameters

Other indications shown within the same display area are:

- **Thrust Limit Display** – Displays the Thrust Limit Mode at thrust lever detent position value, and a fictitious air temperature for flex thrust calculations.
- **Bleed Configuration** – Displays what air is being used for air conditioning, nacelle and wing anti-icing.
- **Idle indication** – Indicates to the crew that both engines are at idle.

Situational awareness – The ECAM display can change colour, shape or outline to highlight unusual or potentially hazardous conditions for example, low oil pressure or high EGT.

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ECAM PRIMARY ENGINE PARAMETERES

ECAM Secondary Engine Parameters

Location

The secondary engine parameters are located on the Secondary Engine (SEN) page of the Electronic Centralised Aircraft Monitoring (ECAM) display on central DU.

Purpose

The purpose of the secondary parameters is to provide an extended indication of engine operation to the crew either automatically during certain phases of the flight or on request by the crew.

Conditions for automatic engine display

When the ENG START rotary selector is selected to IGN / START until the end of the start sequence,

Or

When one engine is in cranking,

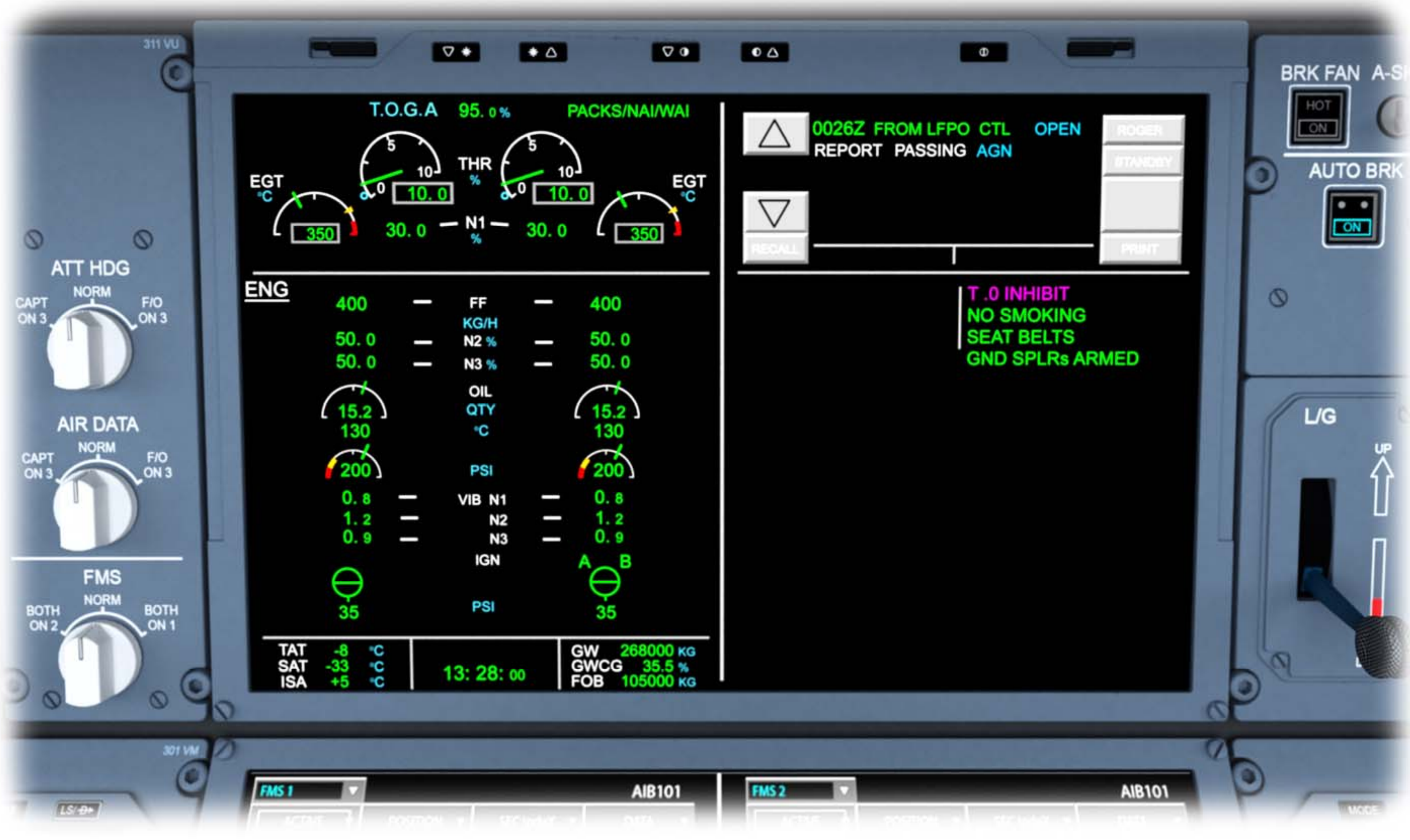
Or

From the setting of take-off power to thrust reduction or 1500 ft/Altitude Above Ground Level (AGL), whichever occurs first.

ECAM Colour coding

The ECAM displays information in various colours. Each colour indicates the importance of the displayed information, or of the failure.

RED	• For configurations or failures requiring immediate action.
AMBER	• For configurations or failures requiring awareness but not immediate action.
GREEN	• For Information in procedure, or in the STATUS page • For checklist items completed by the flight crew. • For memo items
WHITE	• For a procedure completed by the flight crew. • For submenus, condition lines, and titles. • For more information item on the STATUS MORE page. • For a completed deferred procedure title in the checklist menu.
BLUE	• For actions to be completed, limitations to be followed, checklist items to be checked, or for not completed checklists in the checklist menu.
MAGENTA	• For a specific memo (e.g. TO or LDG inhibition).
GRAY	• For checklists completed by the flight crew. • For an action not yet validated by the flight crew (e.g. condition items or a not-sensed procedure that are not activated).



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ECAM SECONDARY ENGINE PARAMETERS

Description

The secondary engine parameters are described below:

Intermediate Pressure Speed (N2) – A digital indication shows the crew the rotational speed of the Intermediate Pressure (IP) System. The N2 speed is shown as a percentage of the maximum N2 speed and is calculated by the EEC.

High Pressure Speed (N3) – A digital indication shows the crew the rotational speed of the High Pressure (HP) System. The N3 speed is shown as a percentage of the maximum N3 speed and is calculated by the EEC.

Fuel Flow – A digital indication shows the crew the fuel flow applicable to an engines operating condition. The indication can be in kilograms per hour (KG/H) or pounds per hour (LBS/H) dependant on operator choice and is calculated by the EEC.

Oil Quantity – Analogue and digital indications show the crew the quantity of oil in the engine oil tank in US Quarts and is measured by the oil quantity transmitter via a conditioning unit to the EEC.

Oil Temperature – A digital indication shows the crew the temperature of the engine oil tank return oil in degrees celsius (°C) and is measured by two Resistance Temperature Devices (RTDs) attached to the lower section of the quantity transmitter. Signals from the RTDs are relayed to the EEC.

Oil pressure – Analogue and digital indications show the crew the differential pressure of the engine oil system. The oil pressure is displayed in in pounds per square inch (PSI). The differential pressure is calculated between the oil pump delivery pressure and the internal gearbox (IGB) scavenge return pressure. Signals are relayed to the EEC.

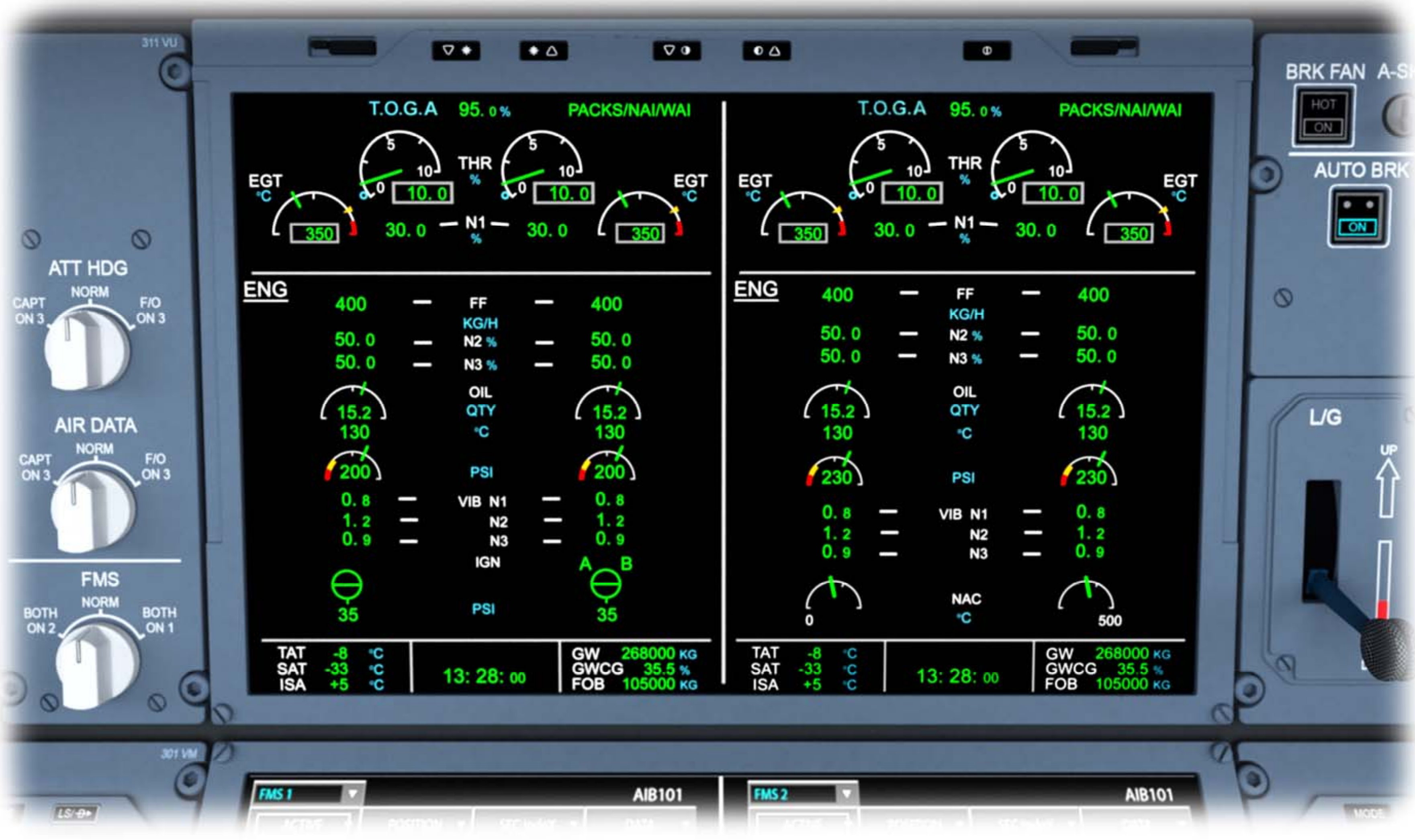
Vibration (VIB) – A digital indication, displays the vibration levels for each of the three rotating systems during engine operation. The indication is in aircraft units (ACU) and is measured by an intermediate casing mounted vibration transducer, which relays vibration levels to the EMU.

Nacelle Temperature (NAC) – Analogue and digital indications show the temperature of the nacelle in degrees celsius (°C) and is measured by the nacelle temperature sensor, located in zone 3. During engine start this indication will not be seen.

Igniter System (IGN) – A digital indication shows the crew which ignition system the EEC has selected during the engine start sequence. The letter A, B or AB defines the system selected.

Starter Air Valve (SAV) – A symbol is shown to the crew to represent the SAV position during the engine start sequence.

Starter Air Duct Pressure – A digital indication shows the crew the air pressure within the starter air ducting in pounds per square inch (PSI).



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ECAM SECONDARY ENGINE PARAMETERS

Pressure and temperature stations

Location

Pressure and temperature stations are located throughout the engine.

Purpose

The purpose of the pressure and temperature stations is to monitor the engine air and gas flows for engine control and engine health monitoring (EHM).

Description

Pressure and temperature stations consists of, pressure and temperature sensors. All engine instrumentation interfaces with either the EEC or EMU by electrical harnesses or pressure lines depending whether the parameter is needed for engine control or engine health monitoring.

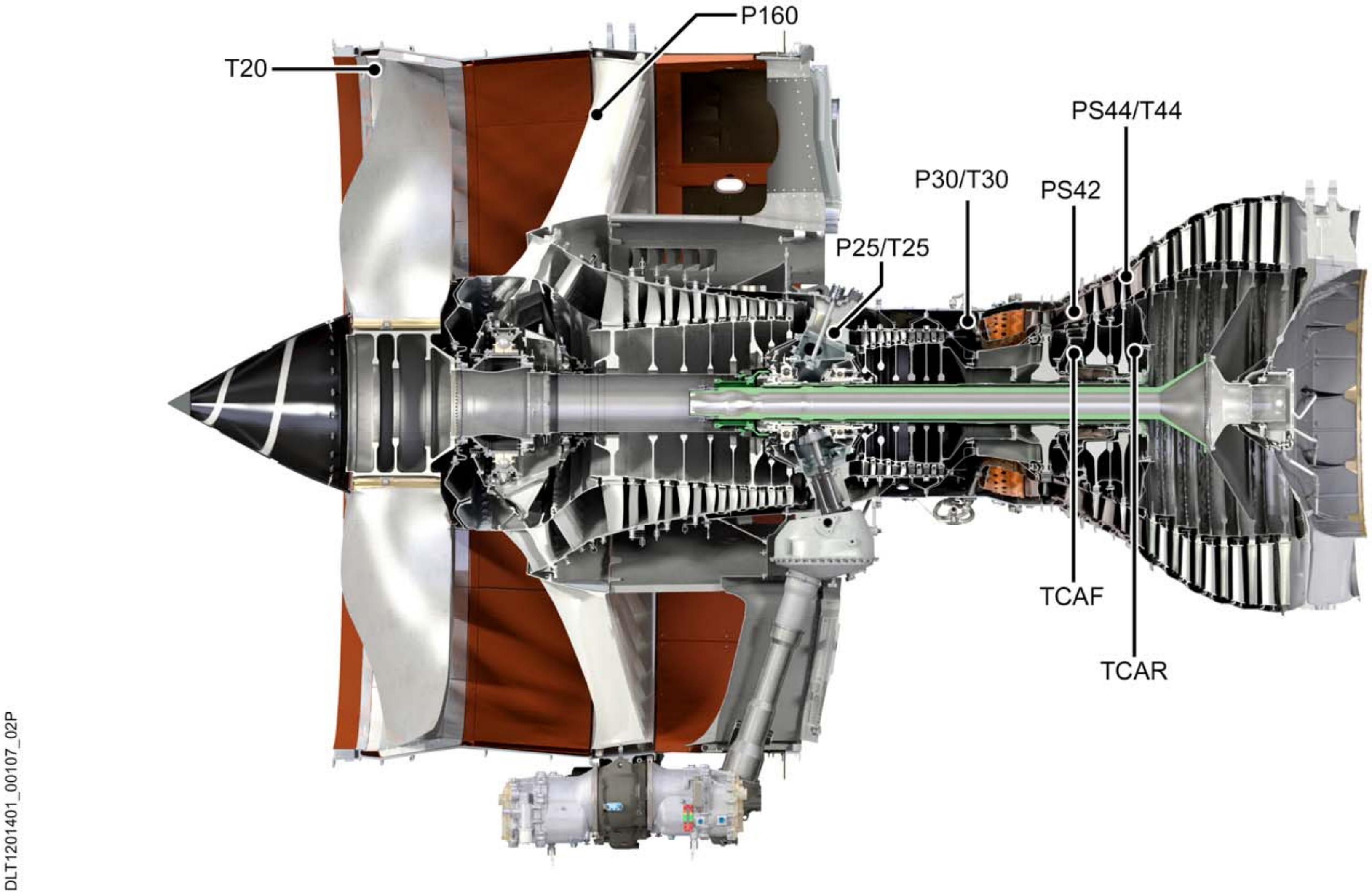
Pressure and temperatures are sensed at different locations, called stations, throughout the engine. A station is a position on the engine where a pressure (P) or temperature (T) or both can be sensed and used for engine control by the EEC and/or engine health monitoring by the EMU.

Engine stations

Specific engine stations are used; these are divided into pressure and temperature sensing. Below is a list of the measurement station numbers.

- Ø = Ambient air (pressure).
- 160 = LP Compressor exit (pressure).
- 20 = Engine intake (temperature).
- 24 = IP Compressor inlet (synthesised temperature by the EEC for ESS anti-ice switching).
- 25 = IP Compressor exit (pressure and temperature).
- 30 = HP Compressor exit (pressure and temperature).
- 42 = IP Turbine inlet (pressure).
- 44 = IP Turbine exit (pressure and temperature).
- TCAF = Turbine Cooling Air Forward (temperature).
- TCAR = Turbine Cooling Air Rear (temperature).

Note: TGT is shown as EGT in the cockpit



PRESSURE AND TEMPERATURE STATIONS

Rotating System Speed Measurement

Location

The LP & IP speed sensors are located around the rotating assemblies. The HP speed is obtained from the PMA rotation. All of these speeds are calculated within each channel of the EEC.

Purpose

The purpose of measuring the speed of the three rotating systems is for engine; control, protection, health monitoring, vibration and indication of engine speed to the crew.

Description

There are three rotating systems in the Trent XWB; they are the Low Pressure (LP), Intermediate Pressure (IP) and High Pressure (HP) systems.

They all rotate independently of each other, and each shaft has its own indication display.

- The LP system is indicated as N1%
- The IP system is indicated as N2%
- The HP system is indicated as N3%

LPC System Speed

There are four LP Compressor speed sensors located in the Front Bearing Housing (FBH), two for each channel. They all interface with the same 60 tooth phonic wheel. The speed sensors are permanent magnets that provide an electrical signal to the EEC / EMU when the teeth of the phonic wheel pass through their magnetic field. One tooth of the phonic wheel is shorter than the rest to give a once per revolution signal to the trim-balance function of the EMU.

IP System Speed

There are four IP Compressor speed sensors located in the FBH, two for each channel which interface with a 60 tooth phonic wheel. The speed sensors consist of permanent magnets that provide an electrical signal to the EEC / EMU when the teeth of the phonic wheel pass through their magnetic field.

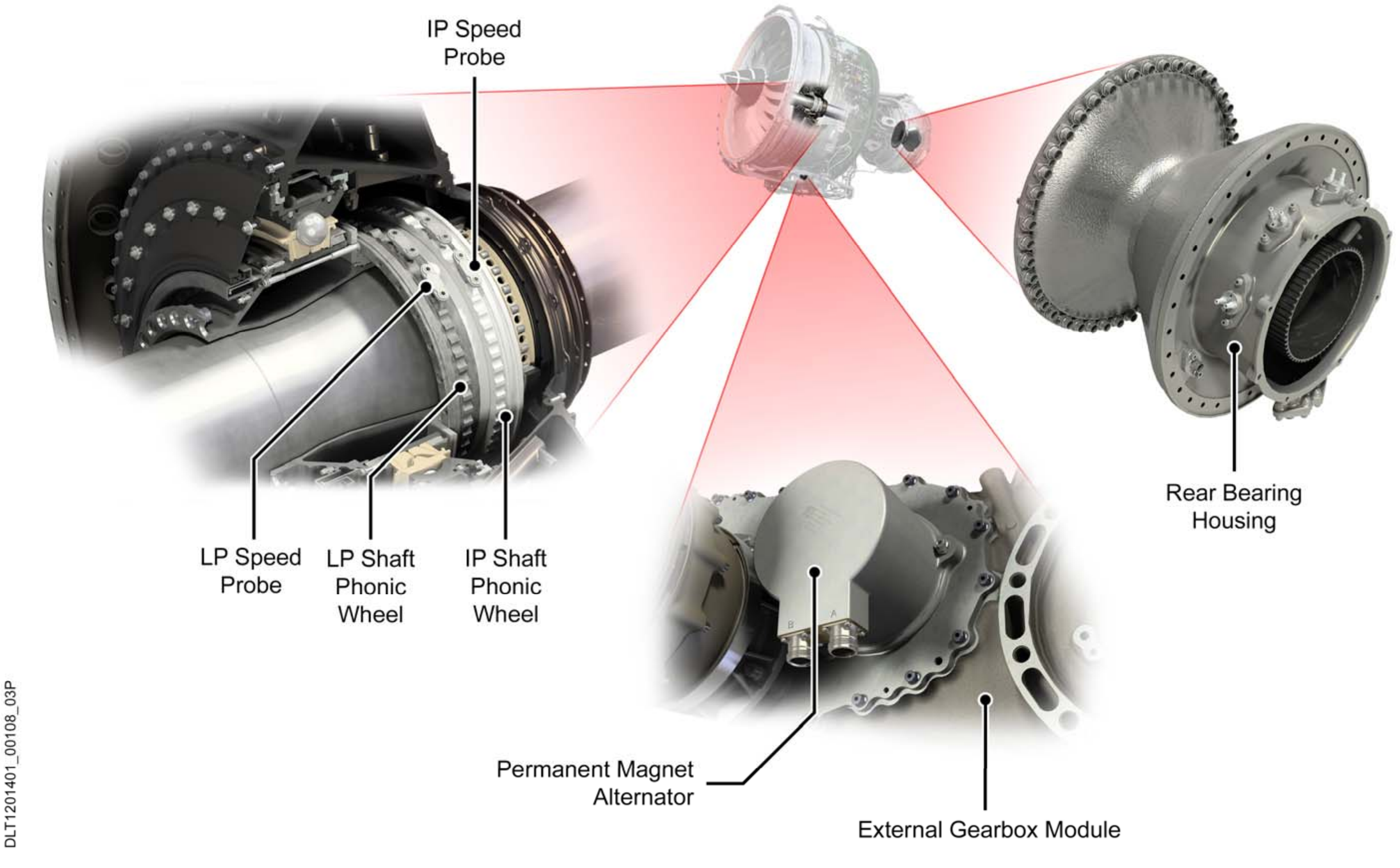
HP System Speed

HP Compressor speed is supplied by the PMA that is turned by the external gearbox and HP system. The EEC / EMU gets HP speed signal using the frequency of an independent single phase winding to channel A and channel B.

LP Turbine speed

LP turbine speed is measured using four permanent magnetic sensors two for each channel and a 60 tooth phonic wheel located in the Tail Bearing Housing (TBH). The EEC uses LP turbine speed for the engine protection systems and for the EMU for engine health monitoring.

The EEC interfaces with each of the speed probes and the PMA by electrical harnesses and provides indications to the flight deck by the AFDX system.



ROTATING SYSTEM SPEED MEASUREMENT

N1 Corrected Thrust Indication System

Introduction

LP speed is the thrust control parameter of the Trent XWB. This is a change from the earlier Trent engines, which used either Engine Pressure Ratio (EPR) or Turbofan Power Ratio (TPR). LP speed was chosen for the XWB because it is a simple, robust measurement suited to high bypass ratio engines.

History

For the earlier Trent engines (500, 700 and 800), hot stream EPR is used as the primary thrust control parameter. EPR has a relatively liner relationship with thrust, a good transient response and is relatively independent of effects of ambient conditions and deterioration. Bypass ratios of later design engines are higher; this reduces the magnitude and range variation of EPR with thrust. The lower hot nozzle pressure ratio of higher bypass ratio engines also makes EPR more sensitive to flight conditions.

TPR was identified as the best replacement for EPR for the Trent 900 and 1000. This parameter effectively measures power available to the LP turbine, and so has a fundamental relation.

LPC speed (known as NL or N1) was chosen as the control parameter for the Trent XWB. It has a good range variation with thrust and is robust, reliable and accurate. The LP speed and thrust relationship is a primary function of a high bypass system and is less sensitive to changes in the core parameters.

Cockpit Indication

The Airbus A350 instrumentation ACUTE (Airbus Cockpit Universal Thrust Emulator) shows thrust as a percentage of maximum rated value.

Cockpit **THR** (Thrust) indication uses corrected N1 values as the thrust control parameter.

N1 Corrected

N1 corrected keeps the ratio of N1 to square root theta a constant, where theta is temperature ratio in absolute units.

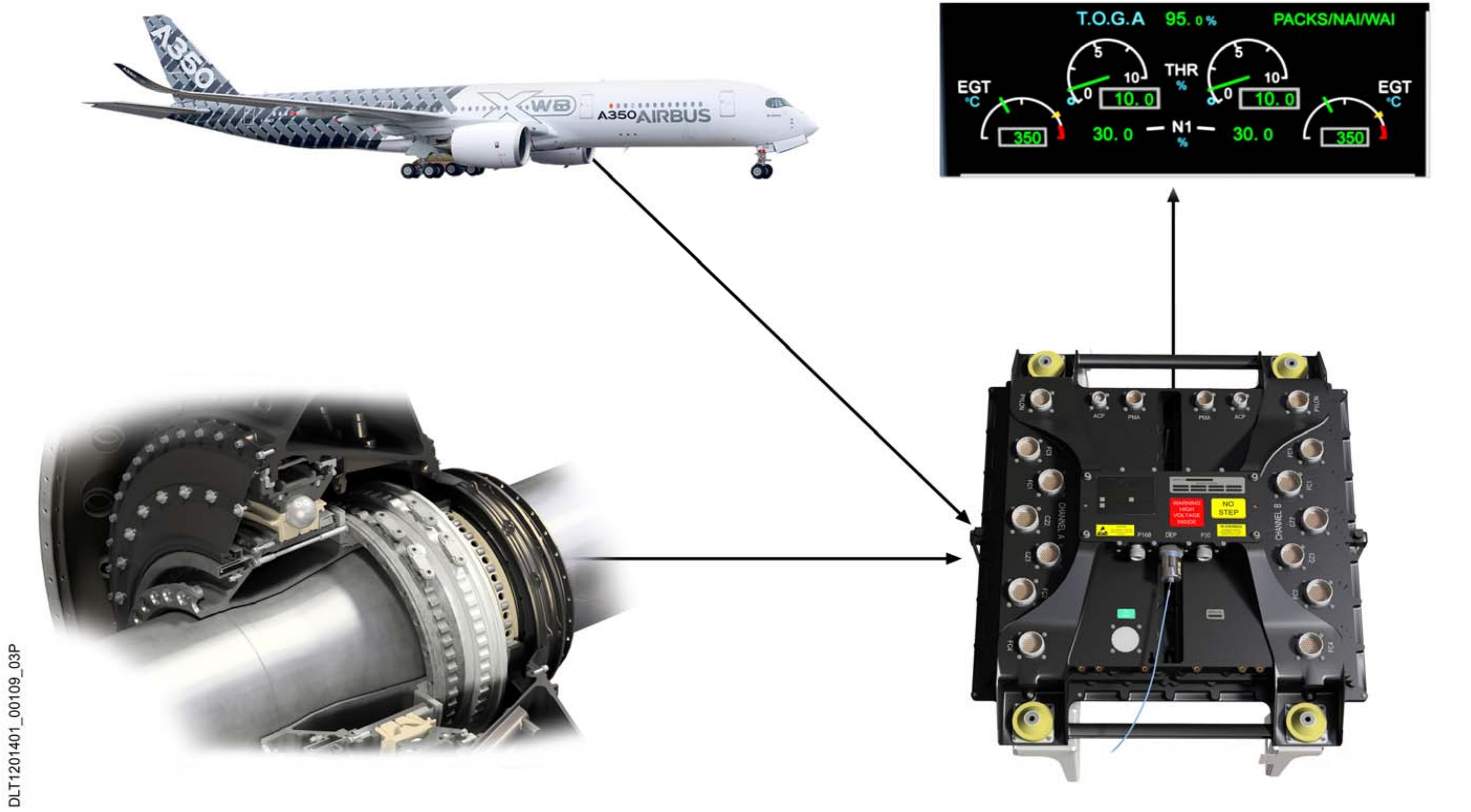
Example: if OAT is -56.5 ° C and N1 is 86.71% what would it have been at 15 ° C on the test bed?

Add 273.15 to °C to obtain °K. Theta is 216.65/288.15 The Square root of theta is 0.8671. If N1√ theta is constant then N1 would have been 100% on the test bed.

Thrust compensation factors

Other factors affecting thrust values:

- Ambient Pressure
- Altitude (ΔP / ΔT)
- Forward air speed (Mach)



N1 CORRECTED THRUST INDICATION

Pressure and Temperature Sensing

Air data parameters are used for calculating environmental and engine conditions such as altitude, Mach number, engine thrust and correction factors for control parameters; used within Engine Control System functions.

Air data parameters are sourced from the aircraft and engine to provide independent sources of air data, which are compared for accuracy and provide redundancy in the event of a fault. The air data parameters used are:

Aircraft (Primary)

- Static Air Pressure
- Total Air Pressure
- Total Air Temperature

Engine (Secondary / Validation)

- Ambient Air pressure (P0)
- Engine Inlet Air Temperature (T20)
- Engine Inlet Pressure (synthesised P20)

The design of the Air Data on the A350 / Trent XWB uses the high integrity communication network to allow a different design from previous applications. The difference is that the aircraft system performs all the comparison and selection of the aircraft and engine air data parameters using all the aircraft and engine sources. The EEC is designed to primarily use valid air data parameters provided by the aircraft when they are available.

Engine Pressure and Temperature Sensors

Ambient Air pressure (P0)

Location

The P0 pressure transducer is located internally to the EEC underneath the fan cowl doors.

Purpose

The purpose of P0 is to provide the EEC with a sense of ambient air pressure (Altitude sensing) for use in controlling the engine thrust.

During normal engine operation the EEC uses aircraft air data. P0 will be used if the EEC cannot validate the aircraft static air pressure.

Description

The EEC measures ambient / static air pressure (P0) using its own dedicated integral pressure transducer that interfaces to the local environment (under the engine cowling) through vent to atmosphere openings in the casing. An independent pressure transducer provides a P0 measurement to each channel. Each channels P0 reading is provided independently to the aircraft as a separate source of P0 for comparison and selection.

Holes in the outer casing allow ambient air to enter the EEC, which is sensed by an internal dual channel transducer.

ENGINE P0 SENSOR WITHIN THE EEC

Engine Inlet Air Temperature (T20)

Location

The T20 probe is located on the inner surface of the engine inlet cowl (TDC) and is accessed via a panel on the outer surface of the Air Inlet.

Purpose

The purpose of the T20 probe is to sense air temperature entering the engine inlet.

Description

The EEC uses T20 values to control; engine thrust, idle settings, bleed valve and VSV schedules.

The temperature sensor inside the T20 probe uses a dual element resistive temperature device (RTD) to supply each channel of the EEC with a value of engine inlet total air temperature.

When the engine is operating the EEC uses aircraft air data Total Air Temperature (TAT). The engine T20 will be used if the EEC cannot validate the aircraft TAT.

IP Compressor Exit Temperature (T25)

Location

The T25 temperature sensor is located in an IP8 cooling air tube at the 2 o'clock position in the intermediate case area.

Purpose

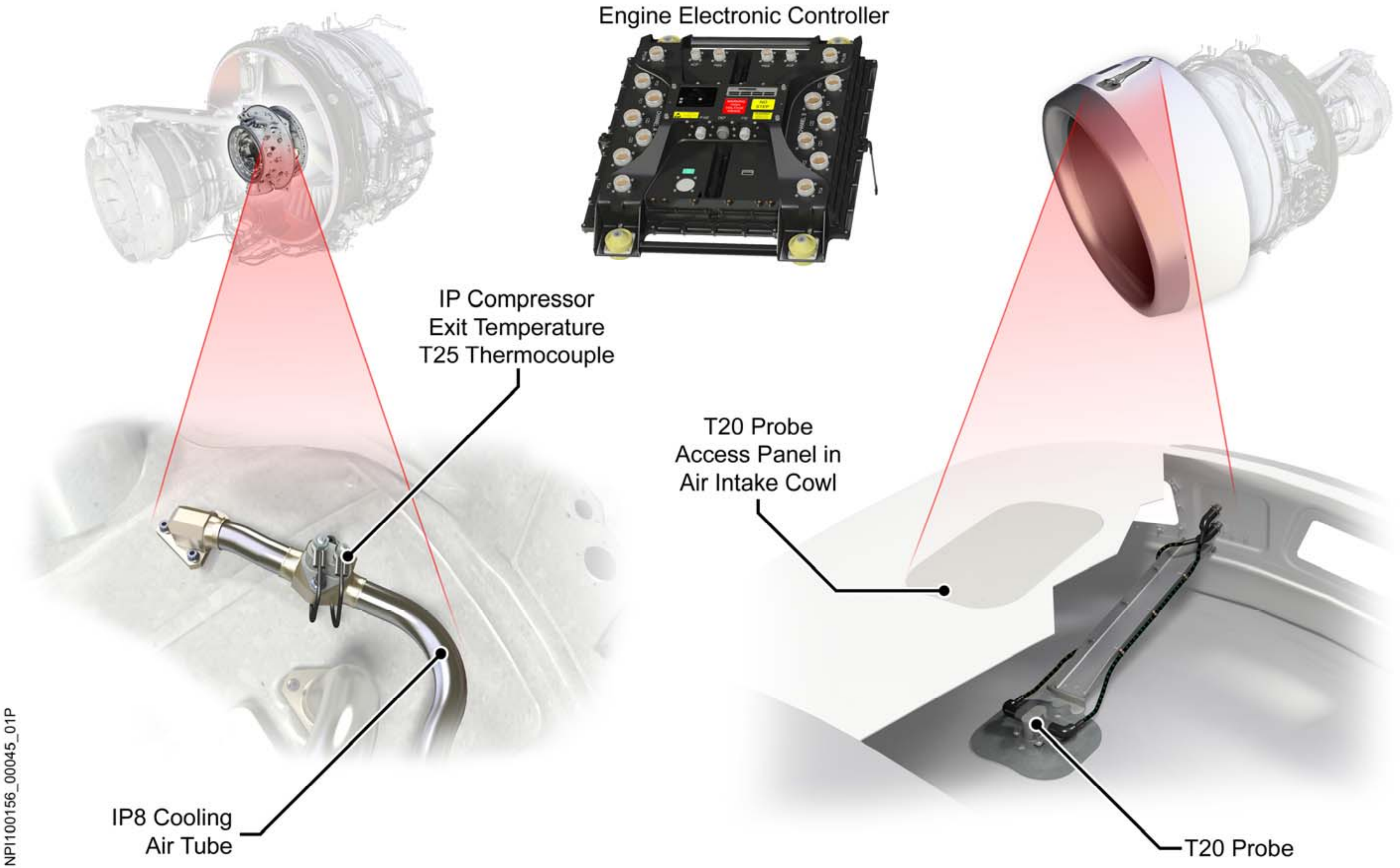
Channel B of the EEC senses T25 for the monitoring of engine health.

Description

The T25 probe is mounted to and protrudes into an IP8 cooling air tube.

The EEC senses the temperature of the IP8 delivery air and reports the temperature to the EMU by the digital data bus that connects the EEC and EMU.

The primary use of the T25 parameter is for engine health monitoring purposes. Where T25 is used for determination of the life of critical, life-limited engine components, and this parameter is used for engine condition monitoring by the EMU.



ENGINE T20 & T25 SENSORS

Engine Pressure and Temperature Sensors

HP Compressor Exit Pressure (P30)

Location

A duplex P30 transducer is located within the pressure brick of the EEC.

Purpose

The purpose of P30 is to provide the EEC with a sense of HP compressor exit pressure for:

- Bleed valve scheduling.
- Thrust setting.
- Engine control.
- Engine start.
- Stall and surge detection.
- Engine Health Monitoring.

Description

A single tube that is just forward of FSN 18 & 19 (11 o'clock ALF) connected to the HP compressor case takes P30 air to the EEC.

The tube is connected to a manifold inside the EEC body that supplies P30 air to a duplex pressure transducer, one for each EEC channel. The tubing between the engine and the EEC has a water trap at its lowest point; designed to accommodate normal amounts of water that will accumulate over many flights due to condensation.

HP Compressor Exit Temperature (T30)

Location

Two simplex thermocouple temperature sensors are located on the combustion case and measure the temperature of the HP compressor air outlet. The thermocouples exit the engine rear of the FSN 2 & 18.

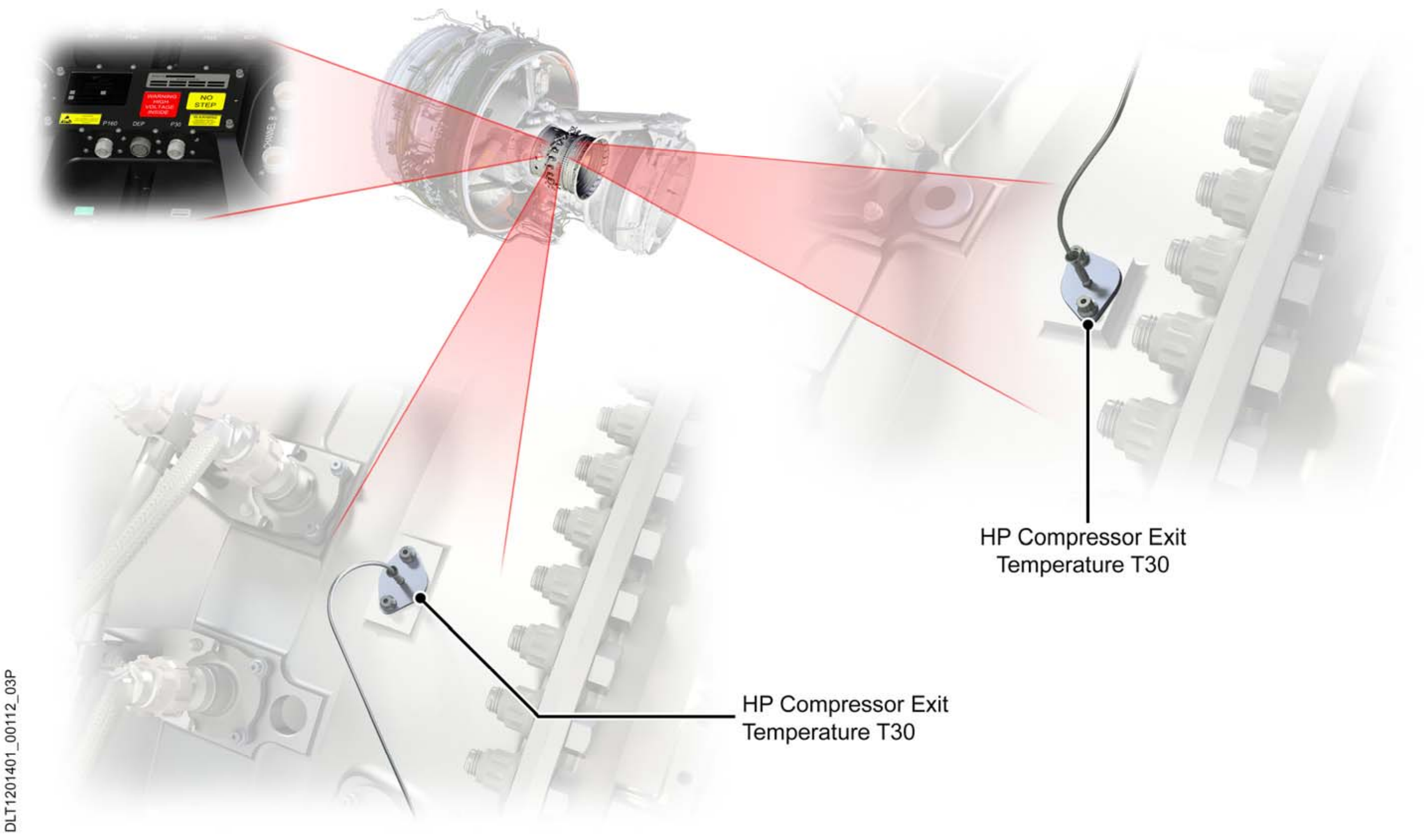
Purpose

The purpose of T30 is to provide the EEC with a sense of HP compressor exit temperature for:

- Engine control.
- Engine start (Stall detection).
- Detection of inclement weather (Flameout).
- Engine Health Monitoring.

Description

Each channel of the EEC receives one T30 thermocouple signal through an electrical harness.



ENGINE P30/T30 SENSORS

Engine Pressures

LP Compressor Exit Static Pressure Ps160 (Fan by-pass air)

Location

The Ps160 sensors are located in the EEC & EMU. The tapping is located in the bypass casing, at the 9 o'clock position.

Purpose

The purpose of Ps160 is to provide the EEC & EMU with a sense of LP compressor exit static pressure for engine health monitoring and secondary thrust calculations.

Description

A single static pressure tapping located at the 9 o'clock position in the rear fancase supplies Ps160 to both the EEC and EMU by a bifurcating tube. The EMU uses this pressure for monitoring turbine performance and the detection of LP compressor damage. The EEC uses this information to improve the accuracy of the P20 model, when the airframe P20 data is unavailable.

IP Compressor Exit pressure (P25)

Location

The P25 sensor is located in the EMU.

Purpose

The purpose of P25 is to provide the EMU with a sense of IP compressor exit pressure for the engine health monitoring.

Description

A single pressure tapping located at the 8 o'clock position in the intermediate case cooling pipe gives P25 to the EMU by a tube for the detection of damage to the IP and HP compressors.

HP Turbine Exit Static Pressure (Ps42)

Location

The Ps42 sensor is located in the EMU.

Purpose

The purpose of Ps42 is to provide the EMU with a sense of IPT inlet static pressure for engine health monitoring.

Description

A pressure tapping within the Turbine Cooling Air Forward (TCAF) thermocouple located at the 1 o'clock position gives Ps42 to the EMU by a tube for monitoring of turbine performance.

IP Turbine Exit Static Pressure (Ps44)

Location

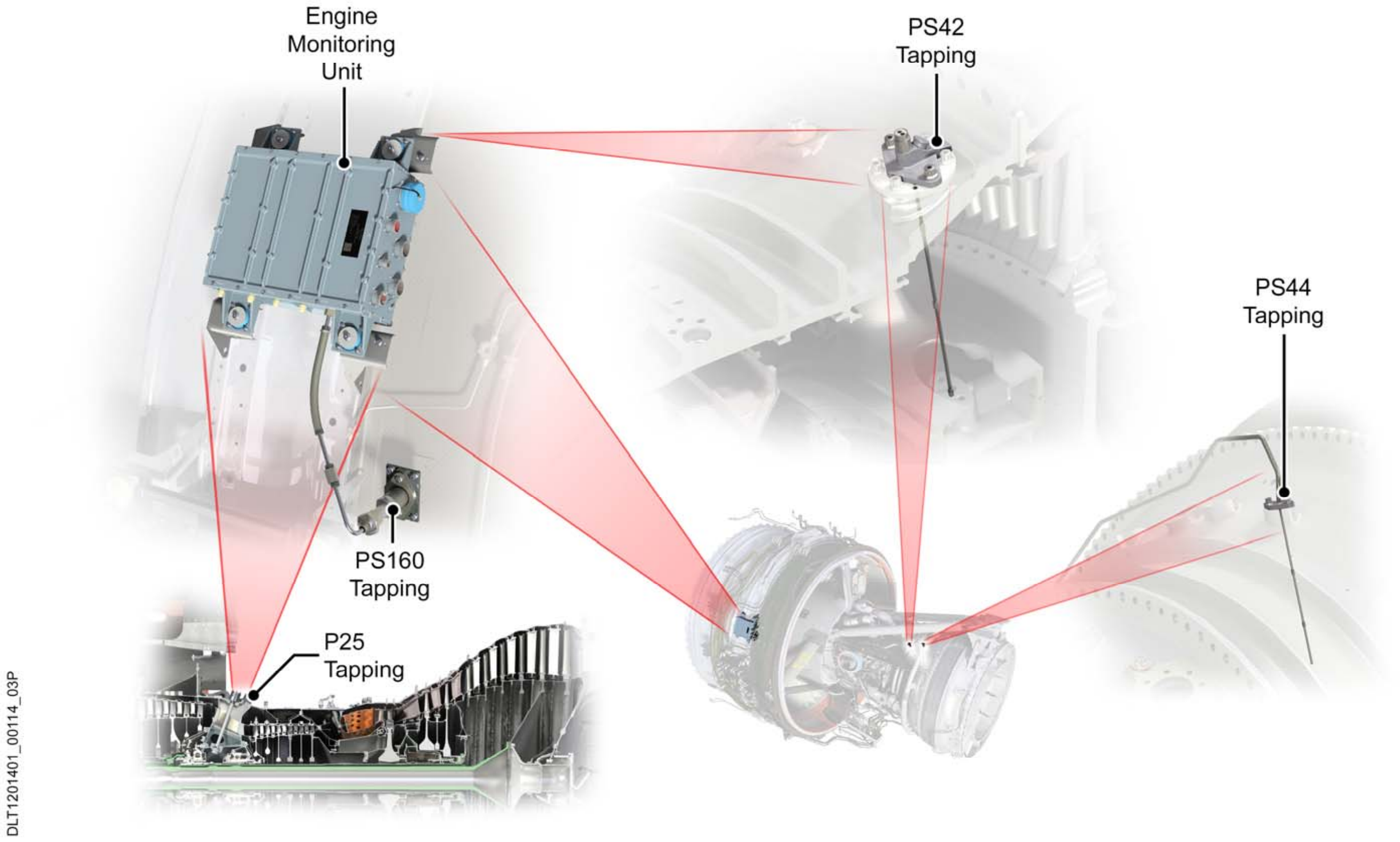
The Ps44 sensor is located in the EMU.

Purpose

The purpose of Ps44 is to provide the EMU with a sense of LPT inlet static pressure.

Description

A pressure tapping on the IP turbine case located at the 11 o'clock position over a single LPT1 NGV gives Ps44 to the EMU by a tube for monitoring turbine performance.



ENGINE PRESSURE SENSORS

Turbine Gas Temperature (TGT) Thermocouple System

Location

The TGT measurement system is located within each channel of the EEC.

The location of the measuring thermocouples is inside the LP turbine stage 1-nozzle guide vanes, which are part of the IPT module.

Purpose

The purpose of the TGT thermocouple system is to measure the Turbine Gas Temperature (TGT) for use in thrust control, engine starting, and flight indication.

Description

12 TGT thermocouples are located within the LPT stage 1 NGVs. Each thermocouple has 2 elements that sense the hot gases going into the LP turbine at different heights. An electrical voltage is generated by the thermocouple that is in proportion to the temperature of the gas passing over the element.

The TGT thermocouples are joined together in two groups of six and connected to channel A or B by a dedicated electrical harness in parallel to the TGT harness by different size connectors.

Each TGT harness has a positive and negative wire, one Nickel Chromium (NiCr) and the other Nickel Aluminium (NiAl), Each harness sends an average value of the raw untrimmed TGT from its six thermocouples to its respective EEC control computer. The EEC applies a trimming logic and indicates the value as Engine Gas Temperature (EGT) on ECAM.

TGT Conversion

The hottest (and therefore limiting) temperature felt by the rotating parts of the engine is HP Turbine Entry Temperature (TET). Since TET is too hot to be measured reliably, except in a test cell, an alternative method is required.

The method used makes a relationship between TET and TGT on the test bed prior to despatch and is known as the ‘TET/TGT’ relationship.

As stated the EGT indication on the ECAM screen is derived from TGT, thus by monitoring EGT the crew can operate the engine to a TGT, without exceeding the engines maximum TET.

Engine TGT Cautions / Warnings

TGT amber limit during starting:

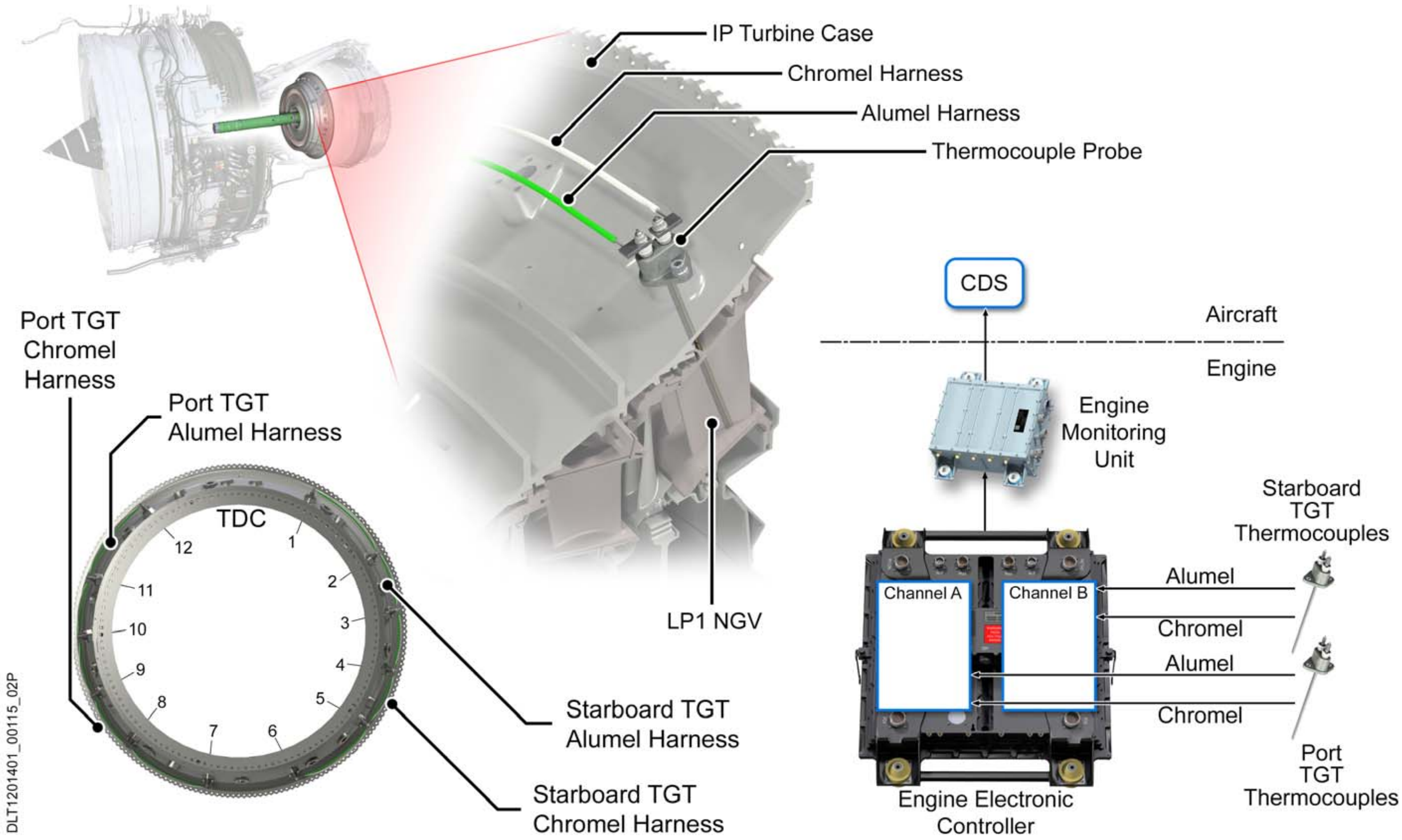
During any ground start the EEC monitors the trimmed TGT value against a ground start amber limit threshold.

TGT amber limit during engine running:

While the engine is running and in-flight the EEC continuously monitors the trimmed TGT value against an engine running amber limit threshold.

TGT red line limit during engine running:

While the engine is running and in-flight the EEC continuously monitors the trimmed TGT value against an engine running red line limit threshold and will detect exceedance if the limit is exceeded for a period.



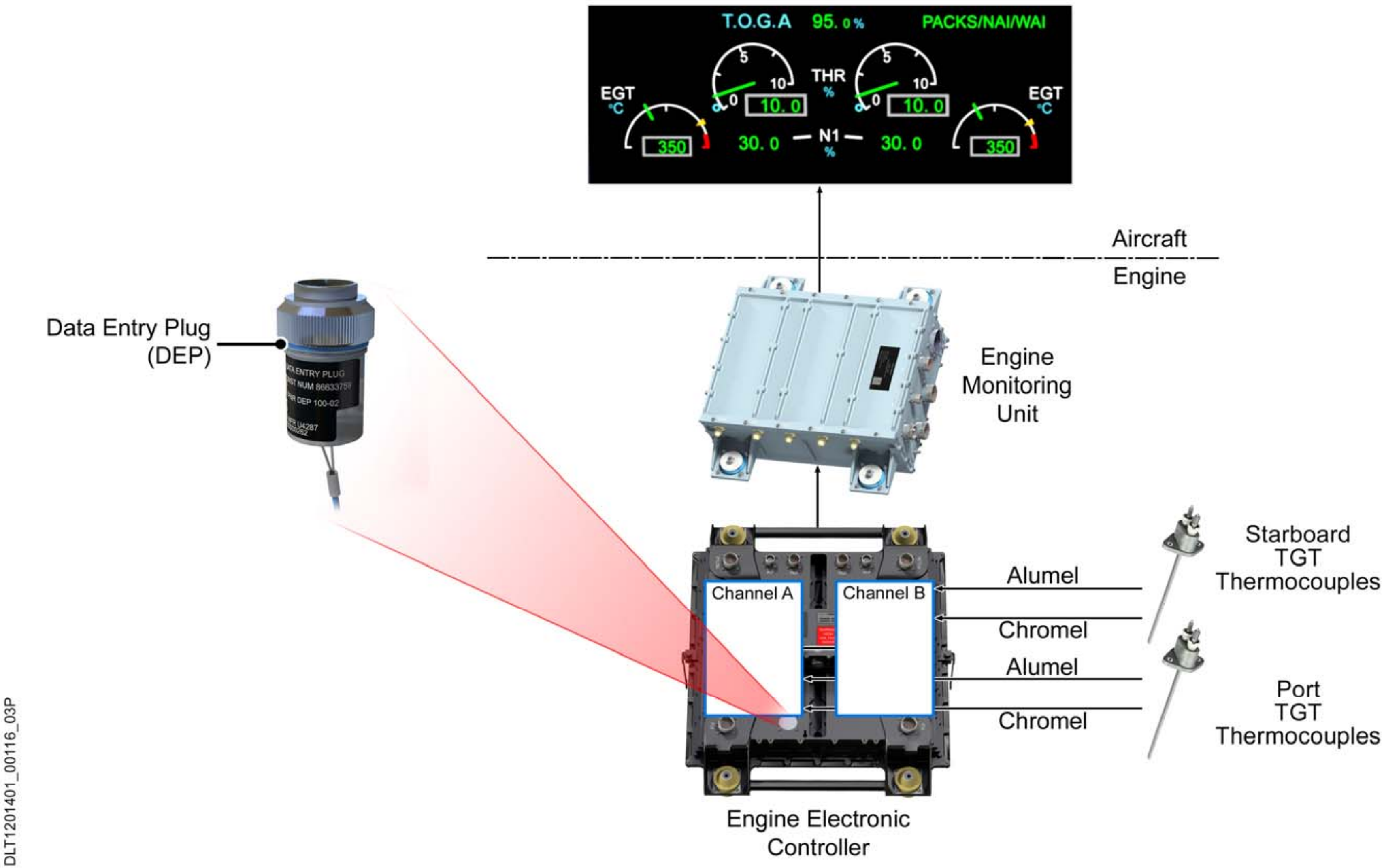
TGT THERMOCOUPLES

TGT Trimming (DEP Input)

To ensure all Trent XWB engines operate to the same EGT limit indication the TGT is trimmed. The amount of trim to be applied is programmed into the DEP after engine test and accessed by the EEC during each power up.

The EEC uses the TGT value to calculate the N1 corrected speed. The N1 corrected is used for engine control as the TGT has a direct relation with the EGT. The EEC sends signals to the EMU and to the flight deck, where the ECAM shows an EGT indication as a value in degrees Celsius.

The EEC uses data held by the DEP to adjust (trim) the TGT signal. This makes all engines operate to the same EGT limit indication shown, as shown on the cockpit EGT display. The quantity of trim that is necessary is programmed into the DEP after engine test and read by the EEC at engine power up.



TURBINE GAS TEMPERATURE (TGT) TRIMMING

Vibration Monitoring System

Location

The vibration measurement system is located within the Engine Monitoring Unit (EMU).

Purpose

The purpose of the vibration monitoring system is to:

- Provide the cockpit with an indication of the state of balance of the three rotating systems (N1, N2 & N3) via the ECAM screen.
- Gather, store and report engine vibration data for the engine health monitoring system.
- Provide data for on-wing trim balancing of the LP compressor.

Description

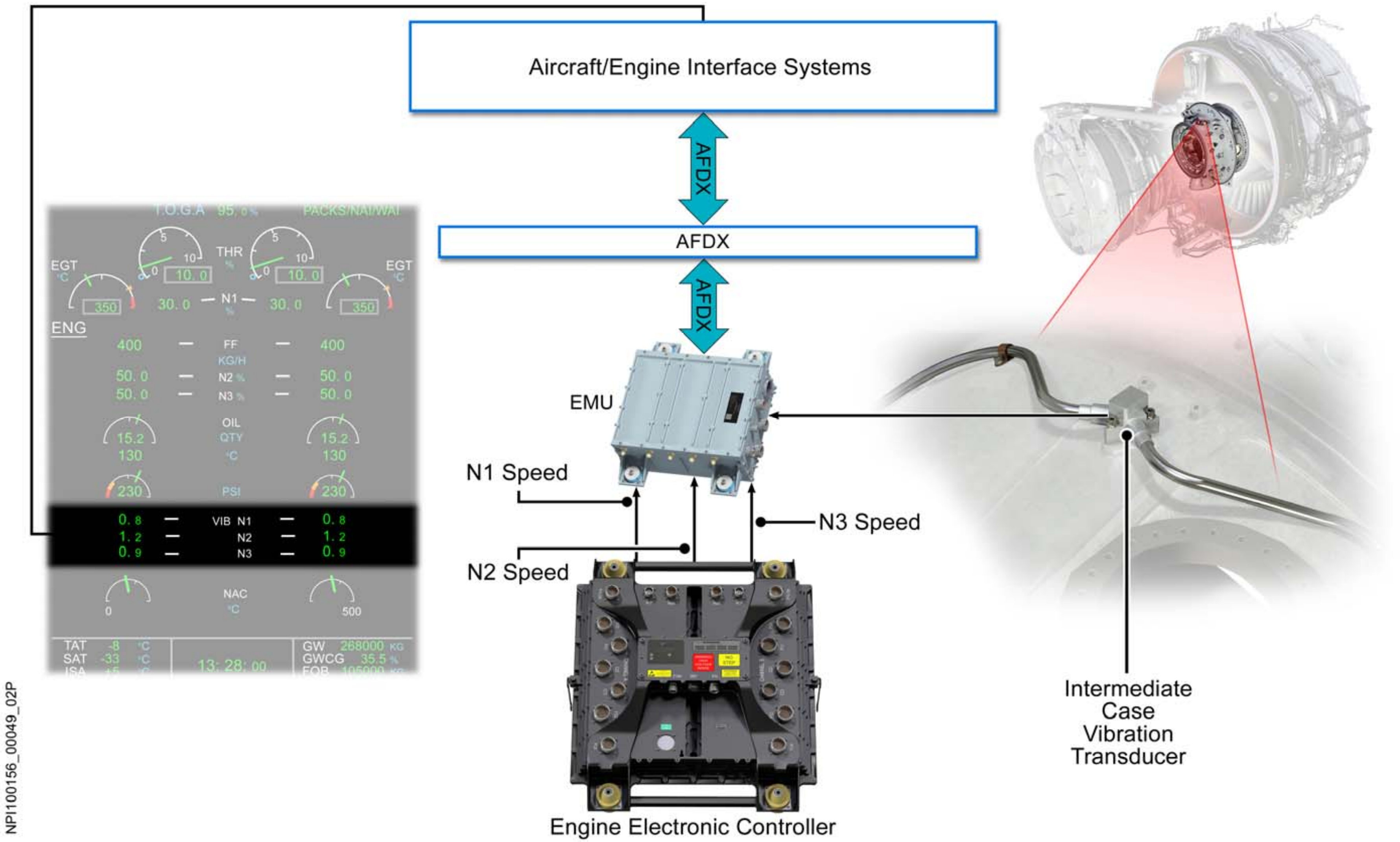
The vibration system components consist of a single transducer, low loss cables, the EMU, analogue connection between the EMU and the EEC and an AFDX connection between the EMU and the aircraft systems.

A single vibration transducer is mounted to the intermediate case and is a dual output (A and B) accelerometer containing two piezo-electric crystal stack elements. As the transducer senses the total vibration in the rotating systems an electrical signal is given to the EMU that is relative to the vibration.

A charge amplifier within the EMU amplifies the signal from the transducer before being processed. The EMU also receives analogue speed signals (N1, N2 and N3) from the EEC. The EMU will align the individual rotating systems vibration to the speed signals and calculate the individual vibration indications that will be seen on the ECAM secondary parameter section.

Fan Trim Balancing

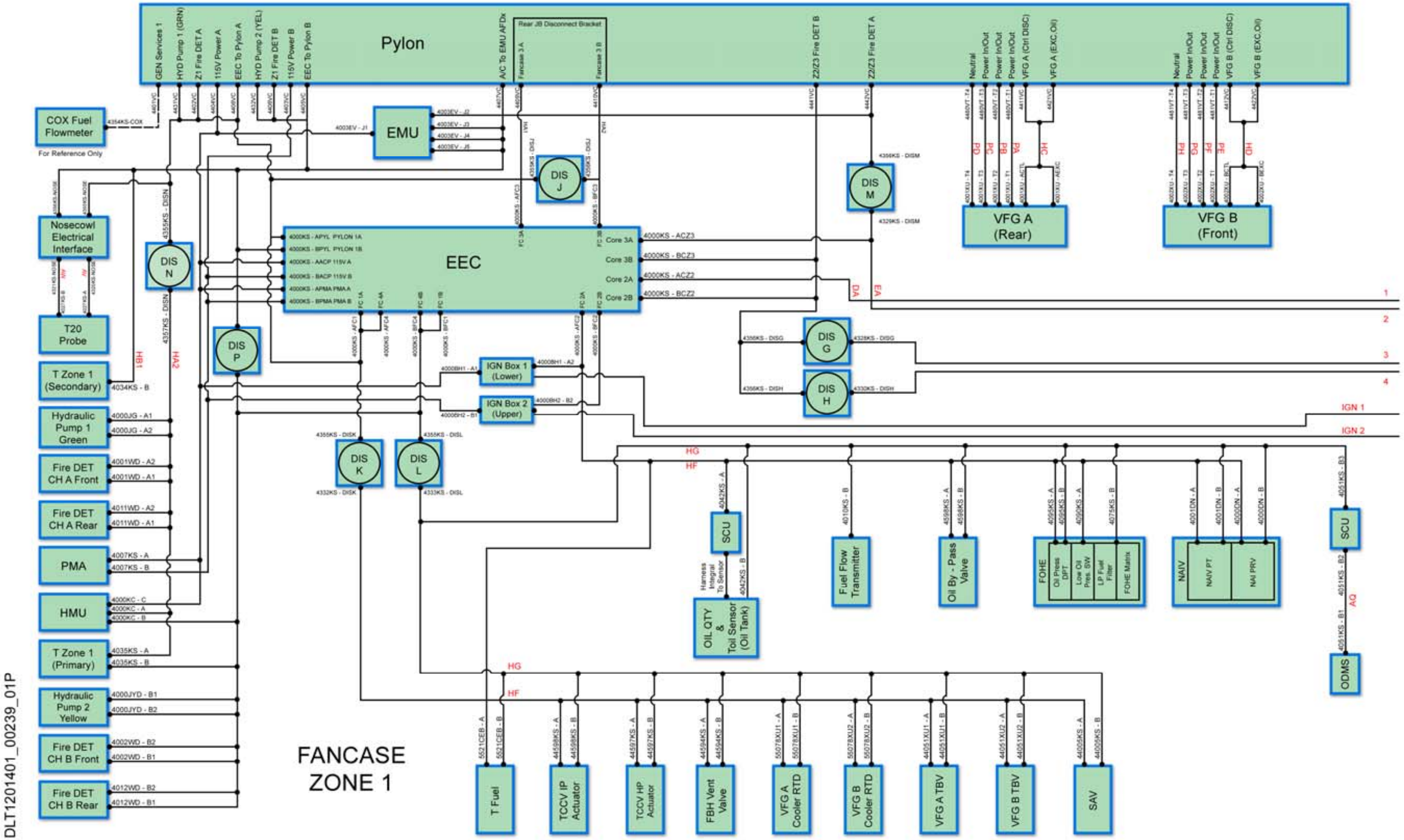
Vibration signals from the single vibration transducer and N1 speed signals from the phonic wheel ‘once per revolution’ tooth are sent to the EMU, which calculates the N1 imbalance and gives a balance weight solution for N1 trim maintenance action.



VIBRATION MONITORING SYSTEM

Engine Indicating Wiring Diagrams
Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMM or FIM electronic documentation.

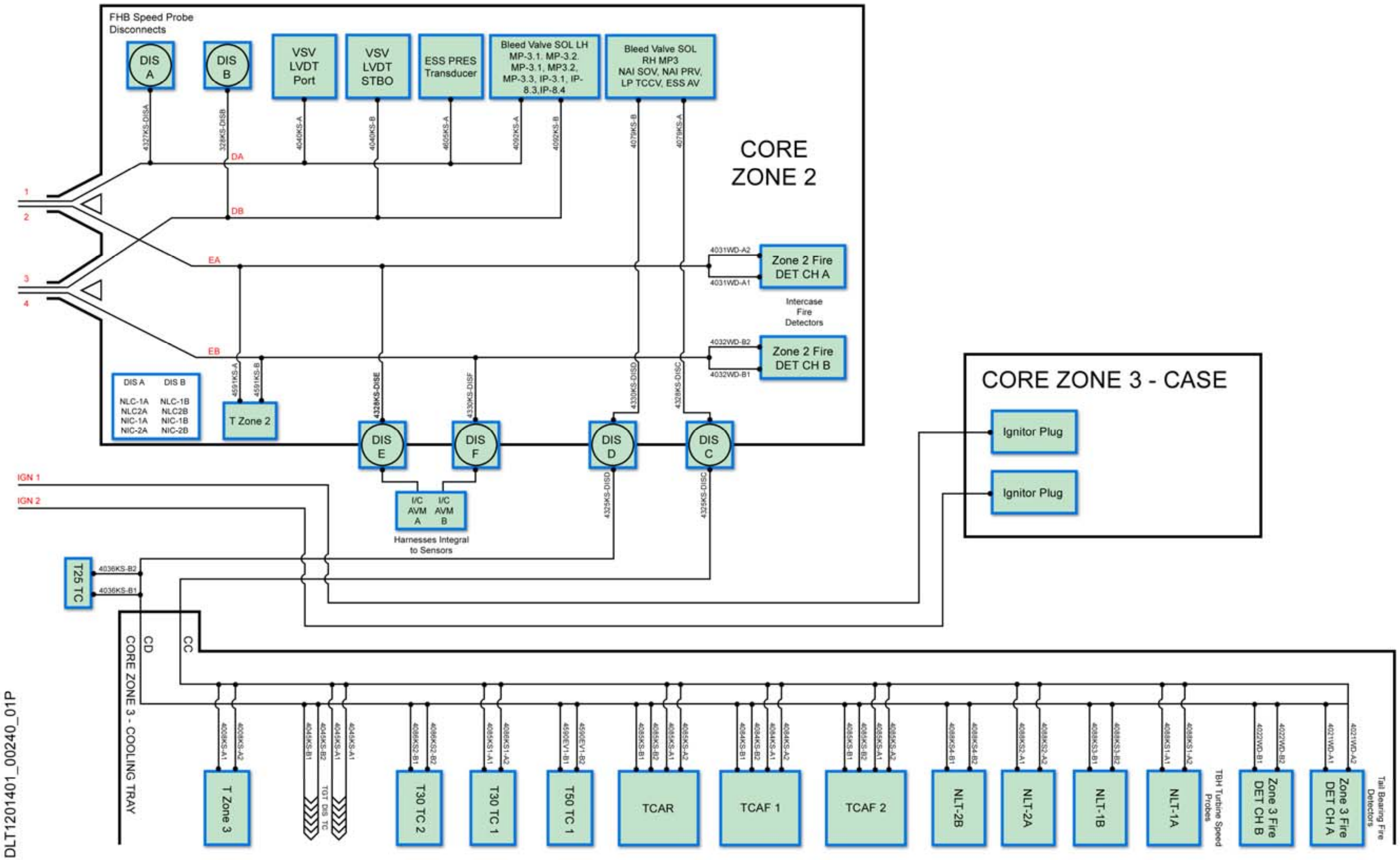


ZONE 1 WIRING DIAGRAM

Engine Indicating Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMM or FIM electronic documentation.



ZONES 2 & 3 WIRING DIAGRAMS

Section- 6 Engine Indicating

Objectives

At the end of the section the student will be able to:

- State the purpose of the engine indicating system.
- Identify the location of the cockpit panels and displays associated with the indicating system.
- Identify the location, purpose and operation of the engine sensing positions.
- Identify the location, purpose and operation of the shaft speed measurement positions.
- State the WARNINGS and CAUTIONS associated with engine indicating system.
- Describe how the Trent XWB engine indicating system interfaces with other engine and aircraft systems.

End of Engine Indicating Section

Section 7 - Oil System

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Section 7 – Engine Oil System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Oil lubrication System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Oil Lubrication System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the engine Oil Lubrication System of the Trent XWB engine.
- State the WARNINGS & CAUTIONS associated with the engine Oil Lubrication system of the Trent XWB engine.
- Describe how the Trent XWB engine Oil Lubrication System interfaces with other engine and aircraft systems.

Oil System Simplified Description

The engine oil system is a full flow re-circulatory system. It supplies oil to lubricate and cool the engine bearings and gears during all operating conditions.

The complete system is divided into three main areas:

- Feed Oil and Cooling.
- Return Oil (Scavenge side).
- Vent, de-aeration and the Breather System.

The oil supply is contained in a tank installed on the right side of the fan case. It incorporates temperature sensors, quantity transmitter, visual sight glass and a gravity oil filling point.

The system is vented through a centrifugal breather located in a housing on the front of the external gearbox.

Oil Cooling

Two Surface Air Oil Heat Exchangers (SAOHE) and a Fuel Oil Heat Exchanger (FOHE) are used to cool the feed oil.

Oil Filtration & Inspection

Pressure, scavenge and line (last chance) filters provide the necessary filtration. Location for Magnetic Chip Detectors (MCD) are provided in each of the scavenge lines returning to the pumps.

Pump Assembly

The pump assembly consists of a pressure pump to move the oil around the system, vent pump to remove oil/air mixture from the Front Bearing Housing (FBH) and nine scavenge pumps to return the oil back to the tank from the following areas:

- Front Bearing Housing (FBH).
- Internal Gearbox (IGB)
- HP Turbine Bearing Chamber (HPT).
- IP Turbine Bearing Chamber (IPT).
- Tail Bearing Housing (TBH).
- Intermediate Gearbox (SAGB).
- External Gearbox (EGB).
- Transfer Gearbox (TGB).
- Breather Scavenge Element (BSE).

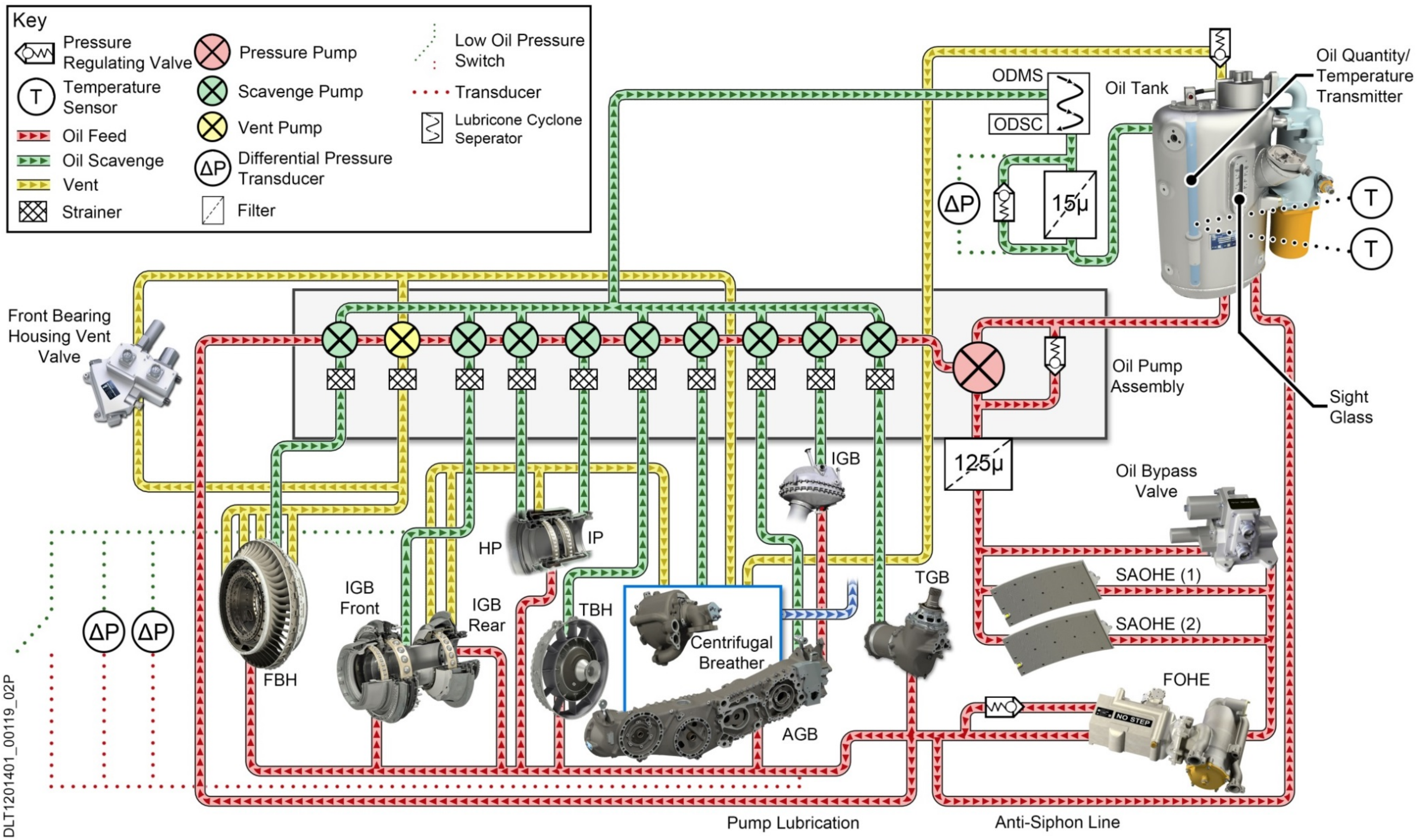
Indications

The following indications are provided on the cockpit:

- Oil Tank Quantity - Secondary Engine Display.
- Oil Temperature - Secondary Engine Display.
- Oil Pressure - Secondary Engine Display.
- Oil Scavenge Filter Blockage - Status Display.

MAINTENANCE TIP

Due to some chemical properties, HTS oil it may stain the gear box external drive areas with a red discolouration. This is a result of an annealing reaction on the gearbox material and is no cause for concern.



OIL – FUNCTION / DESCRIPTION & INTERFACE

Feed Oil, Lubrication & Cooling – Description

Feed oil is drawn from the oil tank by a pressure pump to be circulated through the oil system. A pressure relief valve, set to 635psid, protects the system from excessive pressures due to very cold oil or severe system blockage.

A 125-micron filter cleans the feed oil before entering the system.

The Fuel Oil Heat Exchanger (FOHE) and two SAOHE maintain the oil temperature within limits.

The main function of the FOHE is to transfer heat from the oil to the fuel. A de-congealing valve protects the FOHE core when the engine oil is very cold or if the core is blocked.

From the FOHE the feed oil is supplied through external tubes to the main engine bearings, gears and drives.

The two SAOHE provide additional cooling. When the EEC senses the oil temperature is too high, the EEC will close the oil by-pass valve to allow the oil to flow through the heat exchangers. Each heat exchanger has a de-congealing valve incorporated to provide a by-pass in cases of cold oil or matrix blockage.

RETURN OIL (SCAVENGE)

The return oil/air is scavenged by nine pump elements in the pump assembly from each of the eight lubricated areas of the engine and breather (air oil separator).

Within the pump assembly are positions for eight screw in type Magnetic Chip Detectors (MCDs) to sample return oil from the engine main bearings and gearboxes.

The oil outlet from each of the scavenge pumps join to form a combined scavenge return flow which is sampled by the Oil Debris Sensor (ODS) (part of the Oil Debris Monitoring System)before being passed through the 15 micron Scavenge Filter.

The condition of the scavenge filter is monitored by a differential pressure transducer, which provides a cockpit indication of filter blockage once the aircraft has landed. A by-pass valve inside the scavenge filter will opens at greater than 20psid, allowing oil to flow around the oil system without it being filtered by the scavenge filter.

Two dual channel Resistance Temperature Device (RTD) in the base of the oil quantity transmitter provides the cockpit indication of the oil temperature.

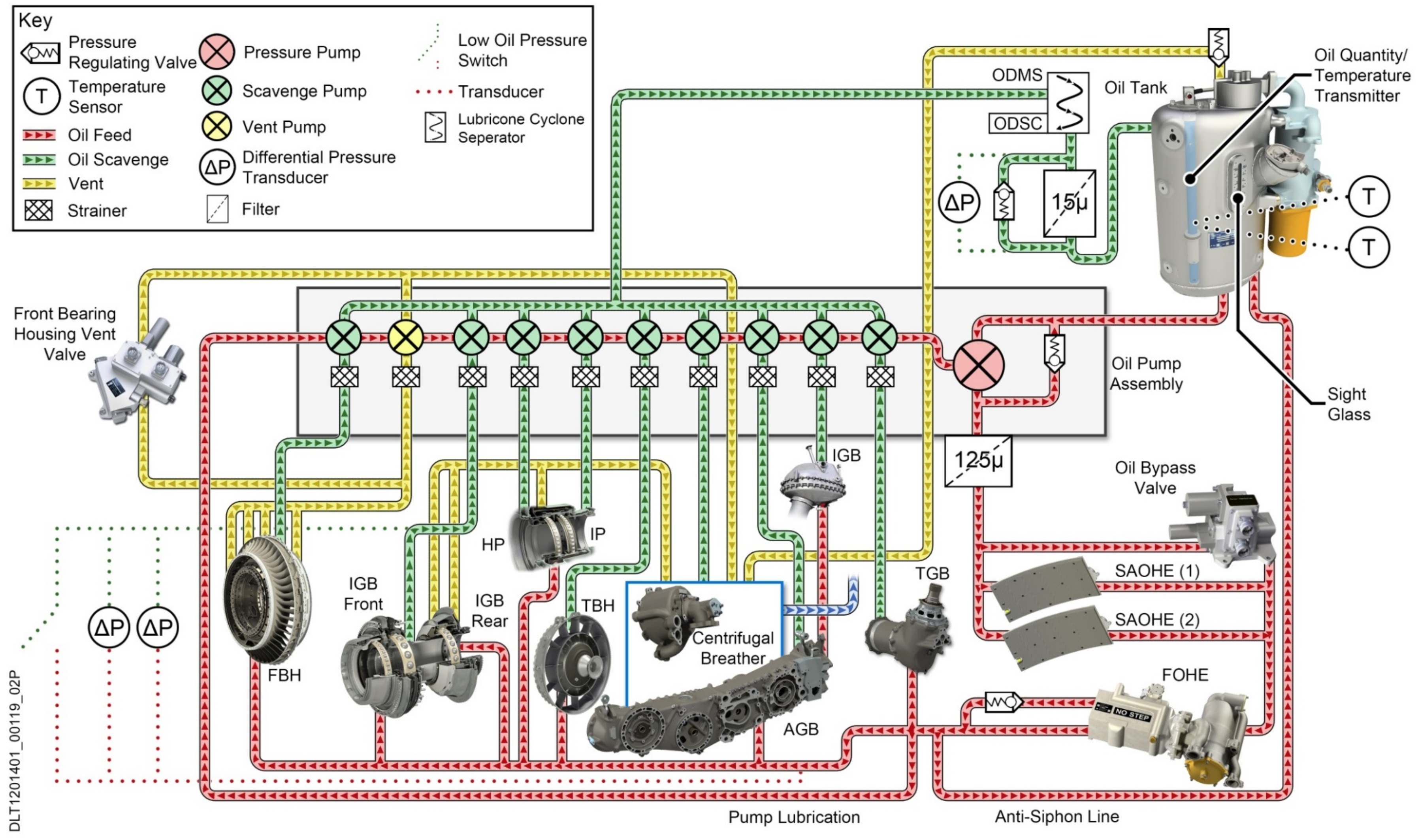
All scavenge oil is de-aerated by a Cyclone Type Separator located within the oil tank.

DE-AERATION, BREATHER AND VENT SYSTEM

IP5, IP8, and HP3, air provides the air pressures required to seal the bearing chambers. The oversized scavenge pumps and vent tubes remove the sealing air that continuously flows through the bearing chambers.

The front bearing housing vent valve (FBHVV) controls the differential pressure in the vent circuit related to the Front Bearing Housing (FBH) compartment. The unit has an inlet and an outlet for scavenge oil/air flow and an inlet and an outlet for actuation fuel flow.

The resulting air/oil mist is transferred to the centrifugal breather where the air and oil is separated. The air is discarded overboard and the oil is scavenged back into the combined scavenge line to the oil tank.



OIL SYSTEM SCHEMATIC

Oil tank

Purpose

Provide a reservoir for the engine oil system.

Location

The oil tank is attached to the fan case on the right hand side.

Features:

The oil tank is an elliptical stainless steel container. It is attached to the Low Pressure (LP) compressor case by three brackets and weighs approximately 13.6 kg (30 lb). The oil tank has these primary items:

- Oil filler and scupper assembly
- De-aerator
- Sight glass and oil measurement scale
- Strainer
- Oil quantity transmitter and two oil temperature sensors

Description

The oil tank provides the reservoir for the engine oil system. A tube at the tank base allows oil to be supplied to the pressure pump to feed the oil system. A coarse strainer at the outlet of the tank prevents debris from damaging the oil pump.

Air is released from the scavenge oil returning to the oil tank by a de-aerator fitted inside the tank. A vent tube passes the air, which still contains some oil, to the centrifugal breather.

The oil filler assembly has a quick release cap and a flapper valve that closes under normal engine oil pressure to minimise oil loss in case the filler cap is not fitted.

Scavenge Filter Housing

Before returning to the oil tank, the scavenge oil / air mixture passes through the scavenge filter which is mounted on the ODMS housing located on the oil tank. The scavenge filter housing contains a 15μ scavenge filter element. The scavenge filter also incorporates a scavenge filter bypass valve which allows the oil flow to be maintained in the event of a scavenge filter blockage. A differential pressure transducer provides the blocked filter alert.

De-Aerator (Part of Oil Tank Assembly)

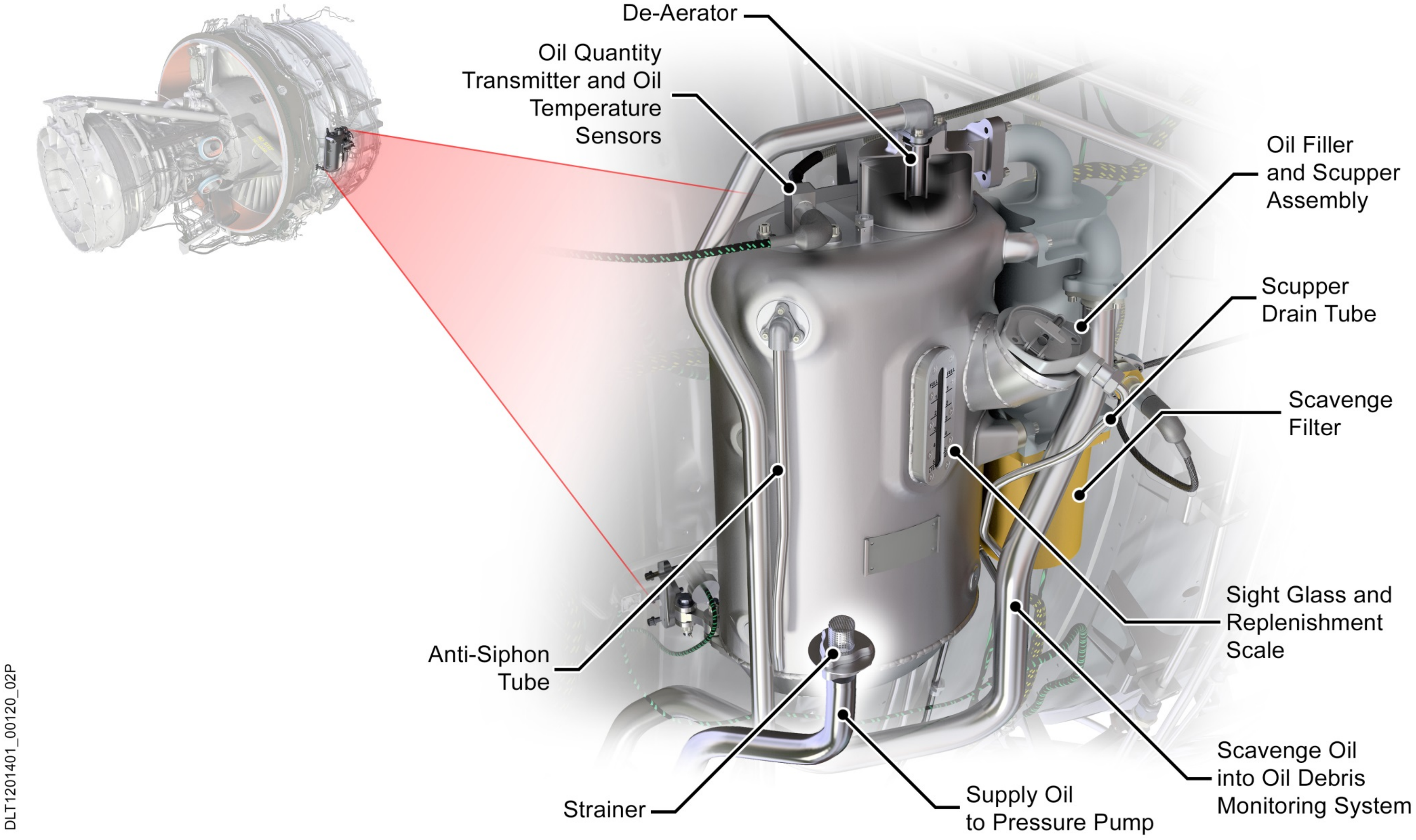
The oil tank accepts scavenge oil / air mixture into a de-aerator in the top of the tank from the scavenge pumps in the oil pump assembly. The de-aerator separates scavenge oil from the air. The air goes out of the de-aerator through a vent tube in the top of the tank to the centrifugal breather on the accessory gearbox. The scavenge oil falls into the oil tank.

Pressure Regulating Valve

The oil tank is pressurised by the oil return line to between 4 and 8 psi a regulating valve is installed in the top of the tank this is to ensure the pressure does not exceed this figure.

Oil Tank Capacities

Oil Tank to FULL line:	22.8 US Quarts; 21.5 litres
Oil tank air space	5 US Quarts


De-Aerator
Oil Quantity Transmitter and Oil Temperature Sensors
Oil Filler and Scupper Assembly
Scupper Drain Tube
Scavenge Filter
Sight Glass and Replenishment Scale
Anti-Siphon Tube
Strainer
Supply Oil to Pressure Pump
Scavenge Oil into Oil Debris Monitoring System
DLT1201401_00120_02P

OIL TANK

Engine Fuel / Oil Heat Management System

Overview

The function of the engine heat management system is to ensure the engine oil and fuel temperature is within specified limits for engine performance. The primary oil cooling is the function of the Fuel Oil Heat Exchanger (FOHE). Two Surface Air Oil Heat Exchangers (SAOHEs) provide secondary cooling when required.

Engine Fuel Oil Heat Exchanger (FOHE)

Location

The engine FOHE is on the right side of the LP compressor case at approximately the 3 o'clock position.

Purpose

The engine FOHE is a two-way heat exchange unit that manages the engine oil and fuel temperatures within specified limits.

Description

The FOHE is the primary method used to control engine oil temperature and is part of the heat management system. The oil system inlet to the engine FOHE is downstream from the pressure oil pump. The unit has an inlet and an outlet for the oil flow and an inlet and an outlet for the fuel flow. The LP fuel filter and related components are also part of the FOHE. An anti-siphon system keeps oil in the engine FOHE at engine shutdown and stabilizes the pressure in the oil system.

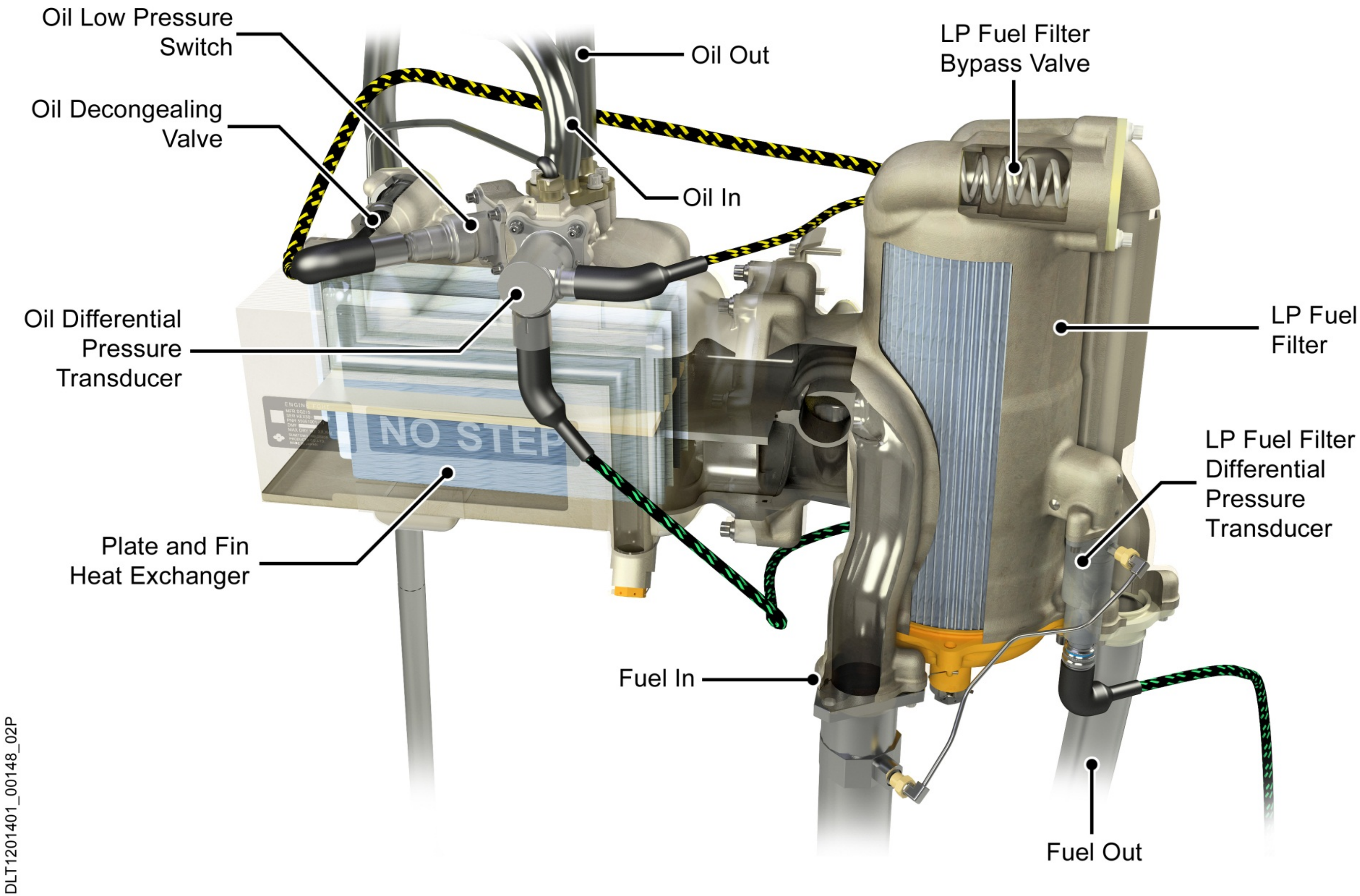
The engine FOHE unit comprises of the following primary items:

- Heat Transfer Matrix
- De-Congealing Valve.
- LP Fuel filter.
- Combined LP fuel filter/FOHE differential pressure Transducer.
- Oil Pressure Differential Pressure Transducer
- Oil Low Pressure Switch.

The heat transfer matrix has plates connected by fins. The oil flows into the unit in one direction and through the plates. Fuel travels in the opposite direction on the outside of the plates and fins. This process allows heat transfer between the oil and fuel.

A spring loaded de-congealing valve in the matrix is operated to allow sufficient oil flow by by-passing the cooling matrix. The valve opens at a specified pressure difference across the FOHE. This lets 60% of the oil flow bypass the matrix without interference when a cold oil condition / blockage occur. Oil and fuel can be drained during maintenance by the removal of a drain plug for each fluid.

The drains plugs are coloured yellow for ease of identification.



FUEL OIL HEAT EXCHANGER (FOHE)

Surface Air Oil Heat Exchangers (SAOHE)

Location

Two SAOHEs are attached to the inside of the rear LP compressor fan case in the upper right hand side quadrant adjacent to the compressor case support struts.

Purpose

The SAOHEs provide additional secondary cooling for the feed oil prior to entering the Fuel Oil Heat Exchanger (FOHE) if required.

Description

The two engine oil SAOHEs are heat exchange units which use LP compressor airflow to cool the engine oil.

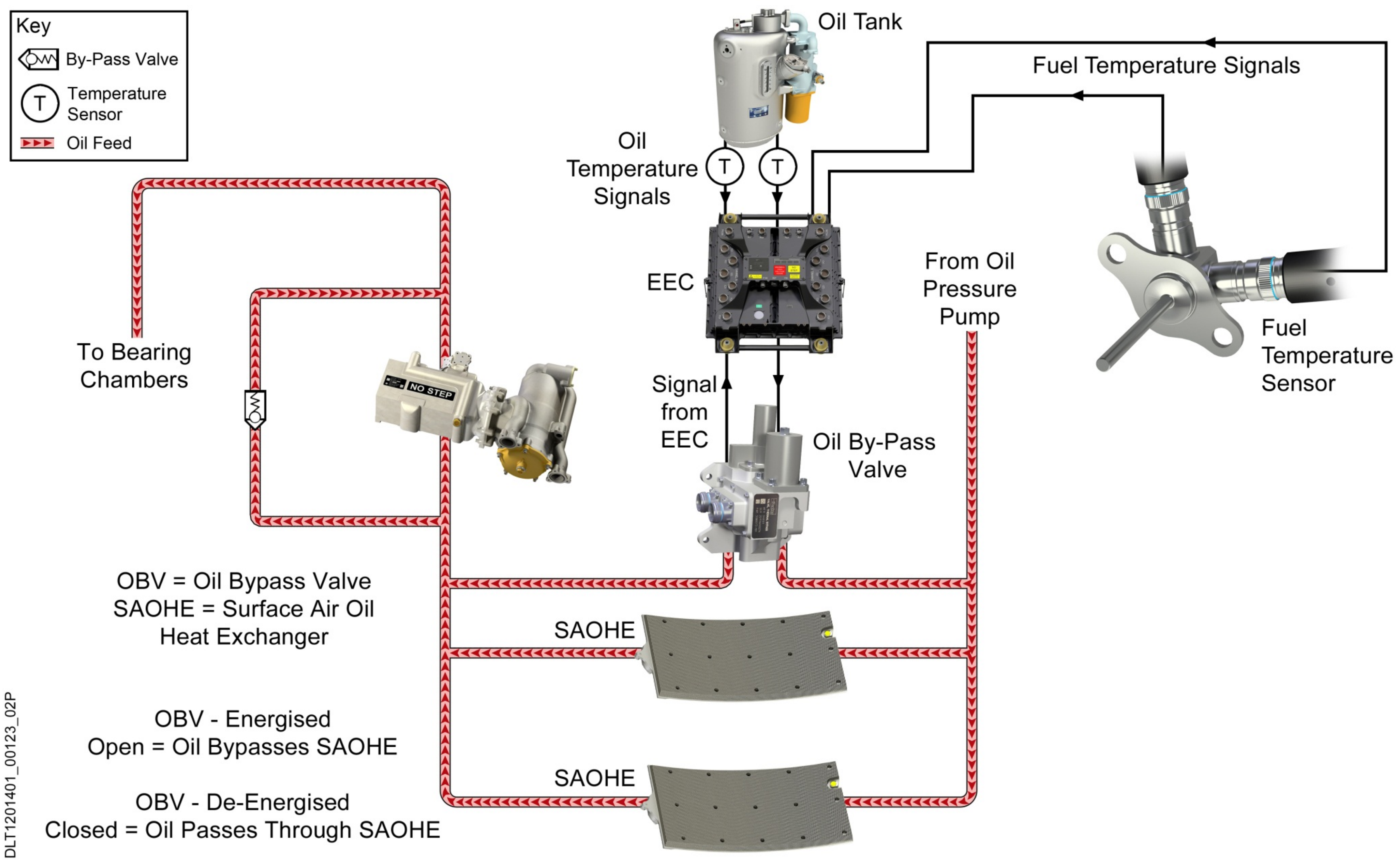
Each unit has a drain plug at the bottom of the heat exchange area for oil removal at maintenance intervals. The SAOHEs are the secondary method used to control engine oil temperature and are part of the heat management system. Each SAOHE has an oil inlet and oil outlet. Full engine oil flow to each unit is permitted by the engine oil bypass valve operation. The oil flows from each engine SAOHE to the engine FOHE.

An engine SAOHE has the following primary items:

- Heat transfer matrix
- De-congealing valve.

The heat transfer matrix has external fins that remove heat from the oil. The inner surface of the heat transfer matrix is above the surface of the adjacent LP compressor case. The air from the LP compressor flows along the fins which allow the heat to transfer from the oil to the air.

The de-congealing valve in each unit gives protection from damage by oil that is too thick or by unit blockage. The valve lets oil flow through the unit without entry into the heat interchange area.



ENGINE OIL HEAT MANAGEMENT SYSTEM

Oil By-Pass Valve (OBV)

Location

On the right side of the LP compressor case approximately the 2 o'clock position.

Purpose

To maintain the engine oil and fuel temperatures within defined limits by switching the SAOHEs in and out of the oil system circuit.

Description

The OBV is controlled by the EEC via a dual channel solenoid in response to oil and fuel temperatures.

The OBV has several functions:

1. To maximise fuel temperature at the FOHE to prevent the fuel filter icing.
2. To minimise fuel heat loss during cruise to aid engine performance.
3. Assist oil cooling.

For control purposes the valve incorporates a LVDT, this signals feedback of the valve position to the EEC.

Operation

The two position oil bypass valve controls the direction of the full oil flow subject to engine oil/fuel temperatures. The unit is part of the heat management system and receives signals from the EEC to control the position of the valve.

The valve position determines oil flow to both engine oil SAOHEs. When energised open Oil flows to the valve inlet from the high pressure oil pump and out of the unit to the engine FOHE, therefore by-passing the SAOHEs.

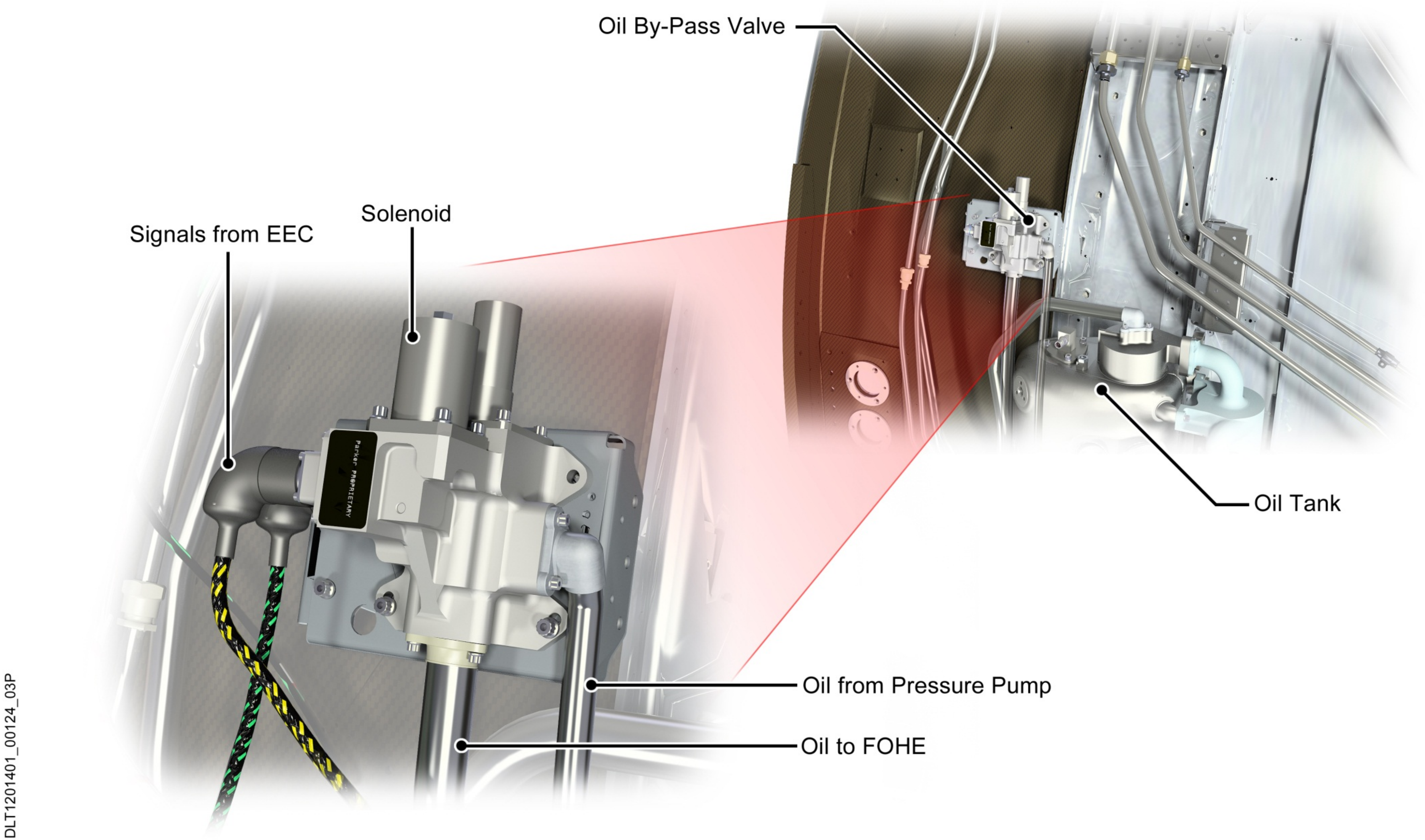
When the engine oil/fuel temperature is above its operating limit, the EEC sends a signal to de-energise the control solenoid in the valve. The valve will close and direct full oil flow to the two engine oil SAOHEs.

The valve energised open approximately 90% of the oil flow through the valve direct to the engine FOHE. The remaining 10% of the oil continues to flow through the two engine oil SAOHEs.

The EEC will close the bypass valve when oil temperatures are greater than 170°c.

The EEC will open the bypass valve when oil temperatures are reduced below 100°c.

The fuel temperature can also effect the operation of the Oil By-pass Valve (OBV) this is explained later in the fuel system section.
The valve is designed to failsafe de-energised closed, which will provide maximum engine oil cooling.



OIL BYPASS VALVE

The Oil Pump Assembly

Location

The oil pump assembly is located on the rear face of the external gearbox.

Purpose

The purpose of the oil pump assembly is to provide a pressurised flow to the oil feed, scavenge and vent systems.

Description

The oil pump assembly consists of:

- A single pressure pump.
- Nine scavenge pumps.
- FBH vent pump.
- High-pressure oil filter.

The oil pump assembly is attached to the rear face of the external gearbox and is driven by a splined shaft. All the pumping elements are located in a common housing that also includes the pressure filter housing.

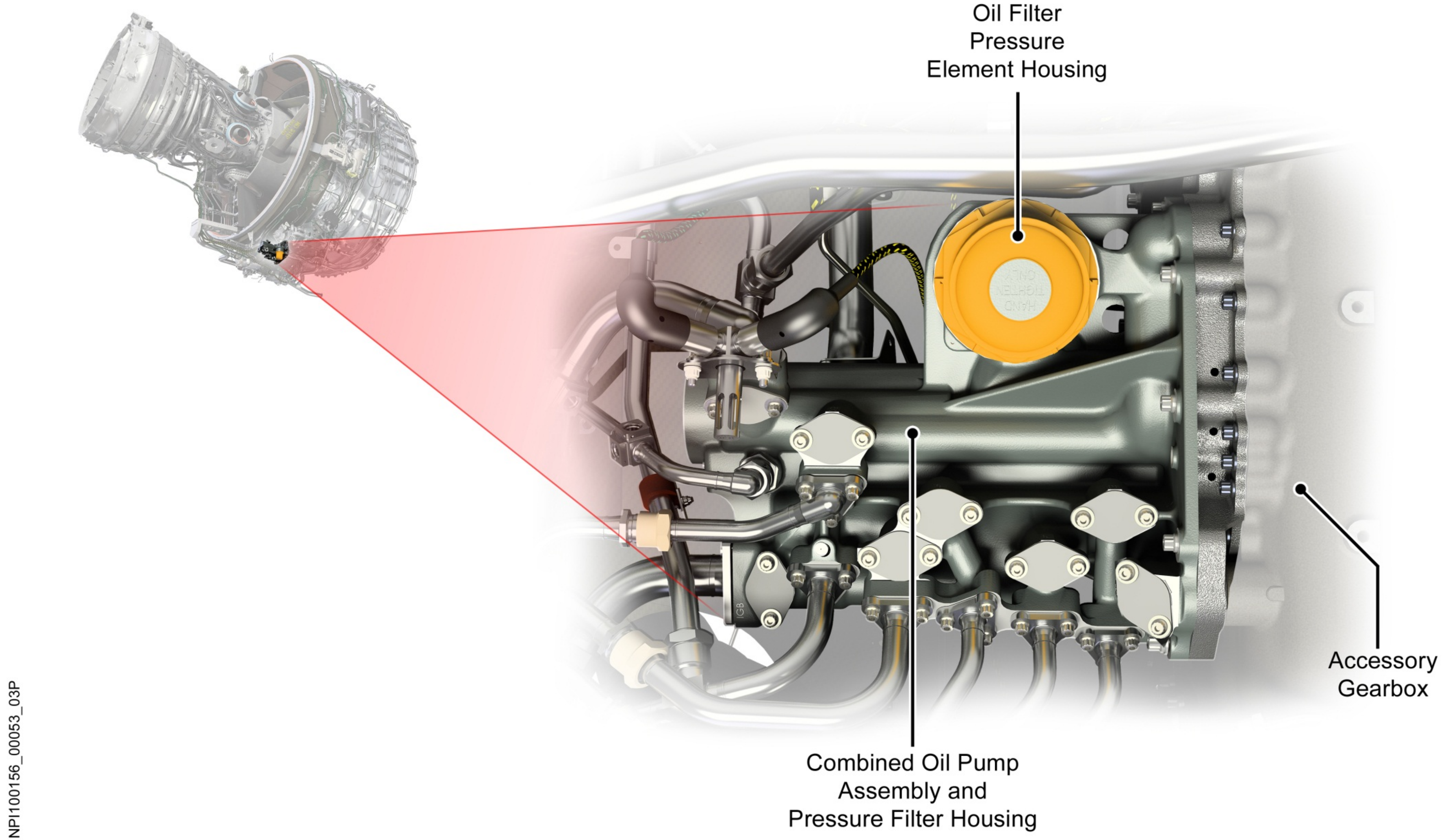
The housing also has provision for eight magnetic chip detectors (MCDs).

The pressure pump has a pressure relief valve to control system pressure and opens if the feed oil pressure exceeds the system limits.

The pressure filter is installed into the pressure filter housing and is covered by a yellow removable cover. A shut-off valve in the housing prevents the loss of oil when the filter is removed during maintenance and an anti-leak valve prevents oil draining back to the oil pump during engine shutdown.

The scavenge pumps remove oil from the bearing chambers and gearboxes. A strainer is located at the inlet of each pump to remove debris. The outlet from each scavenge pump is directed into a single tube to form a common return line to the oil tank.

An additional pump element ensures the sealing pressure in the FBH bearing chamber remains at the correct pressure at low power conditions. It also provides additional lubrication to the pressure pump.



OIL PUMP ASSEMBLY AND PRESSURE FILTER

Return Oil – Scavenge Filter Bypass Valve

Location,

The Scavenge Filter Bypass Valve is located in the ODMS at the base of the Lubricone Unit.

Purpose,

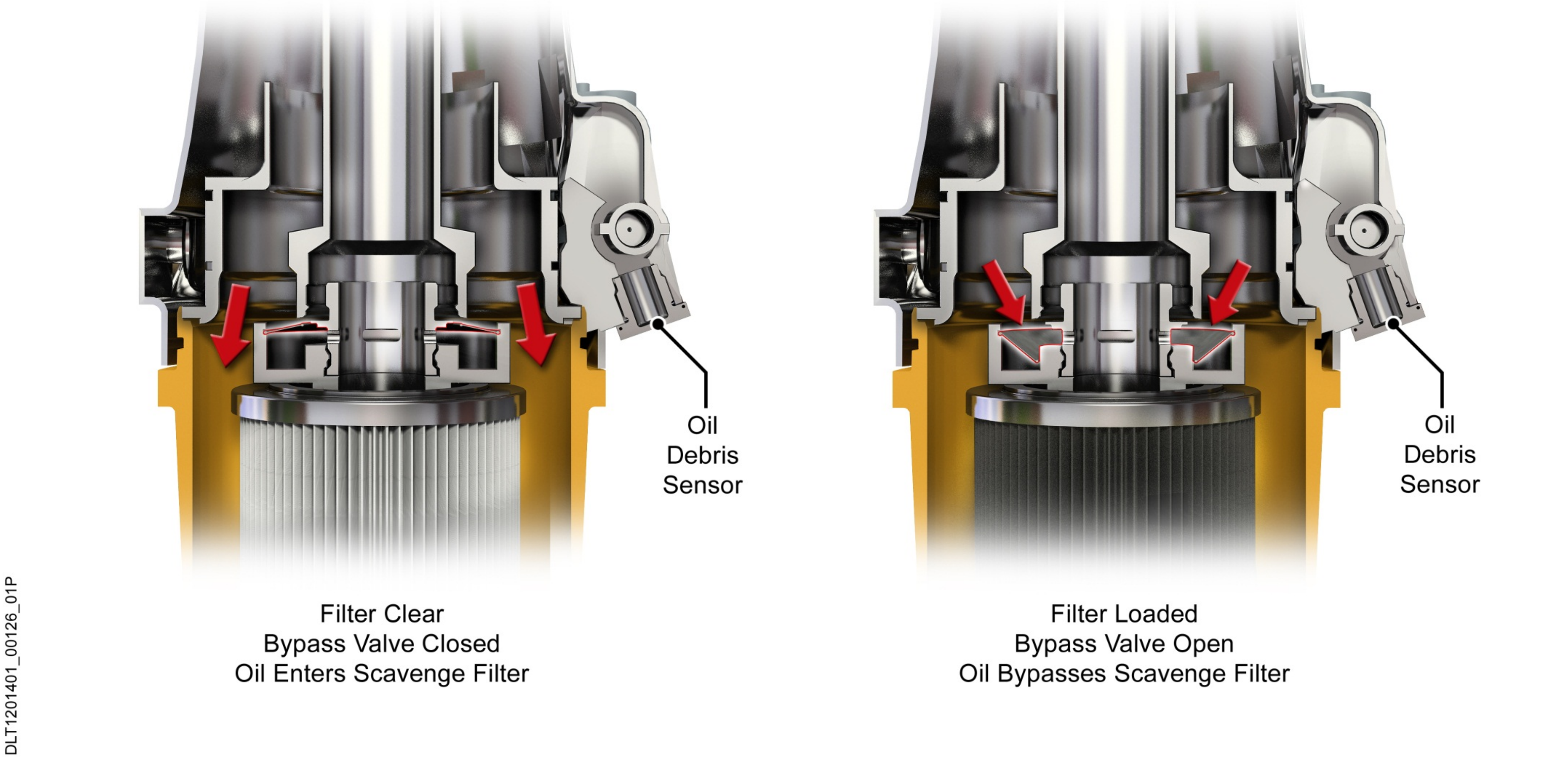
To ensure continuous oil flow when the Scavenge Filter becomes fully blocked.

Description,

If the scavenge filter becomes loaded (Differential Pressure across filter bypass valve is greater than 20 PSID). The Bellville washer deflects, allowing the oil to pass through the filter bypass valve and hence keep the engine lubricated.

The cockpit indication for filter 'clogged' is inhibited in flight, it will only indicate when the aircraft lands.

The 'clogged' message will also appear in the post flight report; this is to alert the ground staff of the 'clogged' filter so that troubleshooting and rectification work can be carried out.



Filter Clear
Bypass Valve Closed
Oil Enters Scavenge Filter

Filter Loaded
Bypass Valve Open
Oil Bypasses Scavenge Filter

OIL SCAVENGE FILTER BYPASS VALVE

Oil Debris Sensor (ODS)

Location

The Oil Debris Sensor (ODS) is installed in a self-closing valve within the separator housing that is part of the ODMS

Description

The ODS is a passive, magnetic, inductive sensor that provides the EEC with a signal that is proportional to the size of the debris.

Before entering the scavenge filter the return oil passes over the ODS. The ODS is located in the separator housing where any ferrous particles are directed onto the ODS for indication to the cockpit and capture in readiness for further analysis. The amount of debris on the sensor causes a signal to be sent to a signal conditioner. The output signal from the signal conditioner is proportional to the size of the debris. Therefore the EEC is made aware of the size of the debris in order to provide the relevant ECAM message to the cockpit.

Note:

During usual engine running only the ODS will be installed. If debris is found on the ODS then the diagnostic MCDs will be installed on the oil pump to isolate the source of the debris.

Diagnostic Magnetic Chip Detectors (MCDs)

Location

The eight Magnetic Chip Detectors (MCD) are located in the scavenge lines of the oil pump assembly.

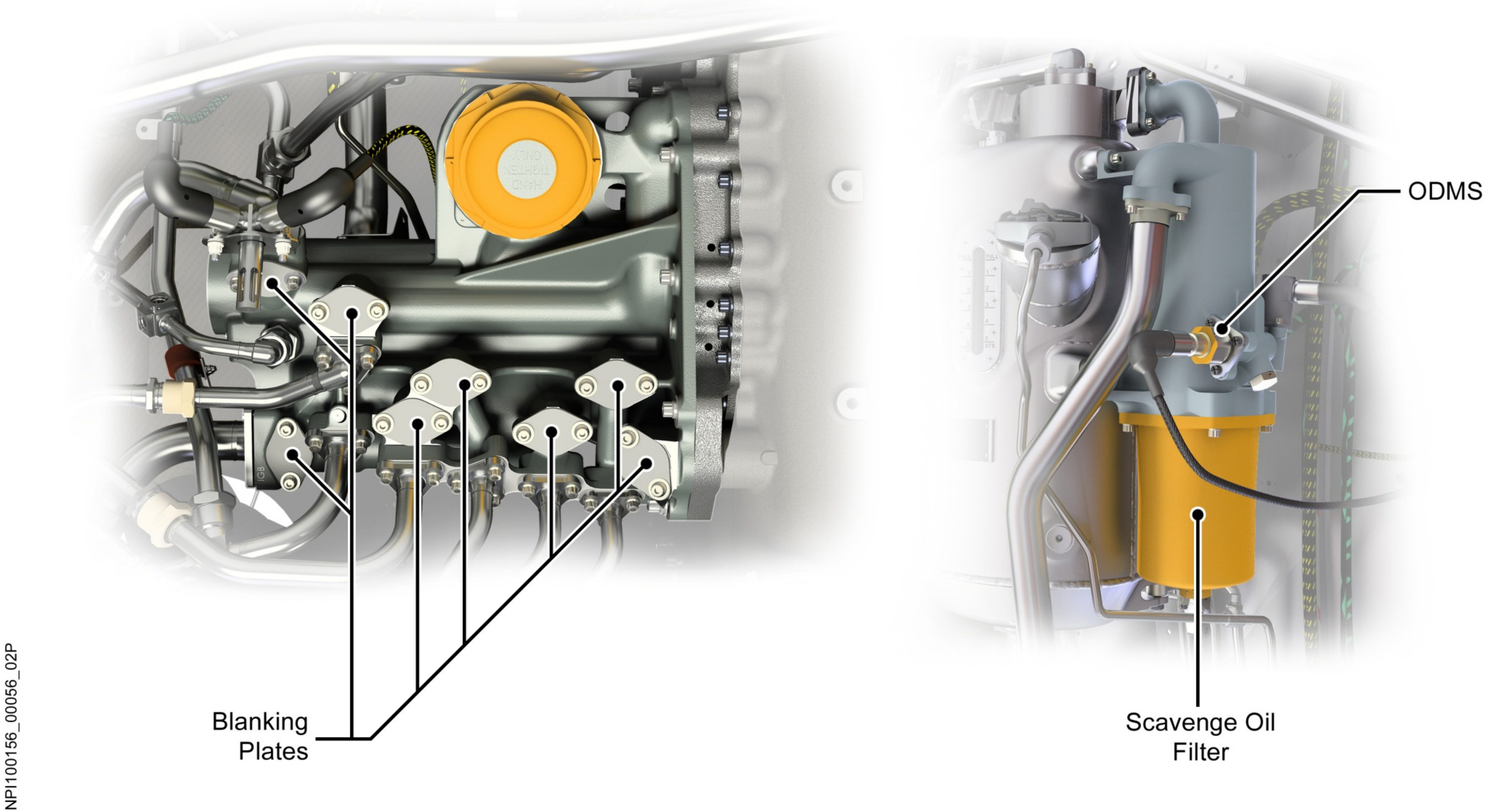
Description

The screw-in MCD assembly consists of a housing and Magnetic Chip Detector, which has a magnetic end. When the MCD assembly is installed the magnetic end is located into the scavenge oil flow so that any ferrous debris attaches itself to the magnet.

To prevent oil leakage when an MCD is removed the housing has a self-closing valve.

The MCDs collect debris from the bearing housings and gearboxes and are identified as follows:

- Front Bearing Housing (FBH).
- Internal Gearbox (IGB)
- HP Turbine Bearing Chamber (HPT).
- IP Turbine Bearing Chamber (IPT).
- Tail Bearing Housing (TBH).
- Intermediate Gearbox (SAGB).
- External Gearbox (EGB).
- Transfer Gearbox (TGB).
- Breather Scavenge Element (BSE).



DIAGNOSTIC MCD AND OIL DEBRIS SENSOR

Oil Debris Monitoring System (ODMS)

Introduction

Monitoring of the debris within the oil system allows for early detection of impending failures of bearings, gears, splines and other oil lubricated components.

Location

The Oil Debris Monitoring System (ODMS) is located and attached to the right side of the oil tank. The system also includes other units that are part of this system and attached to the right side of the fancase.

Description

The ODMS consists of the following units:

- Lubricone Unit.
- Oil debris sensor (ODS).
- Oil debris Signal Conditioner (ODSC).
- Engine Electronic Controller (EEC).
- Engine Monitoring Unit (EMU).

Operation

Oil enters the ODMS separator housing from the Scavenge Pumps via a Combined Oil Scavenge Line on the right side of the unit and travels through the Lubricone unit. The Lubricone unit induces a swirl into the oil flow allowing debris within the oil to be centrifuged into a cavity where it will be analysed by the Oil Debris Sensor (ODS).

Having passed through the Lubricone the oil flows over the bypass valve, through the scavenge filter and into the oil tank.

Oil Debris Sensor (ODS)

The ODS is a passive, magnetic, inductive sensor that provides the EEC with a signal that is proportional to the size of the debris.

Oil Debris Signal Conditioner (ODSC)

A signal from the ODS is received by the Debris Signal Conditioner. The signal is then amplified, filtered and compared to pre-determined threshold levels before being passed to the EEC.

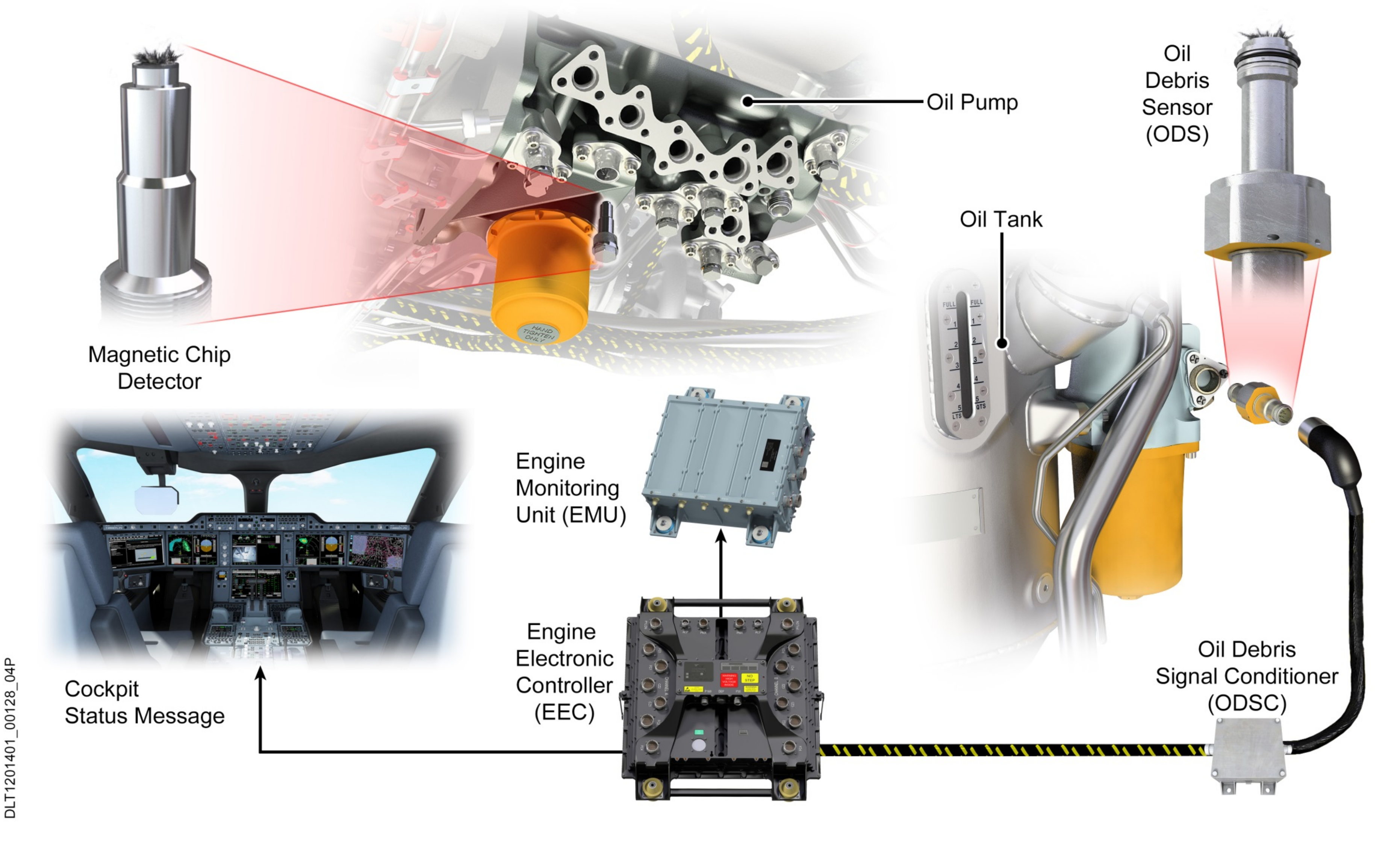
- A 10-millisecond signal is output to the EEC if a small’ particle is detected.
- A 20-millisecond signal is output to the EEC if a ‘large’ particle is detected.

Engine Electronic Controller

Channel A provides the 15V DC required to power the ODSC and also counts the number of small and large particles detected depending on size, quantity and rate of occurrence. Depending on the amount of debris detected the EEC will set an ECAM status message as required.

Engine Monitoring Unit (EMU)

The EMU will receive ODS information from the EEC and will use the information as part of the Engine Health Monitoring suite of software to aid detection and minimize customer disruption.



OIL DEBRIS MONITORING SYSTEM (ODMS)

Oil Indications

Oil Quantity/Temperature Transmitter

Location

Mounted on the oil tank upper surface

Purpose

To provide the EEC cockpit indications of oil quantity/temperature within the engine oil tank.

Description

The oil tank has an oil quantity transmitter for electrical oil quantity indication; the transmitter consists of two concentric tubes, which form the plates of a capacitor with the engine oil acting as the dielectric, as the oil level in the tank changes so does the effective dielectric value, which causes a measurable change in capacitance. The transmitter signals are relayed using channel ‘A’ only via a signal conditioning unit (SCU) to the EEC.

The transmitter also incorporates two oil temperature sensors one for channel ‘A’ and one for channel ‘B, they are dual channel Resistance Temperature Device (RTD) giving oil temperature indication to both channels of the EEC before processing the signal for the cockpit display via the CDS. Channel ‘A’ is wired via a SCU and Channel ‘B’ is directly connected to the EEC. As well cockpit indication oil temperature signals are used for starting and fuel scheduling during acceleration.

The EEC continuously monitors the outputs of the transmitters and in the event of a sensor failure the EEC selects the remaining alternate channel.

Sight glass

Location

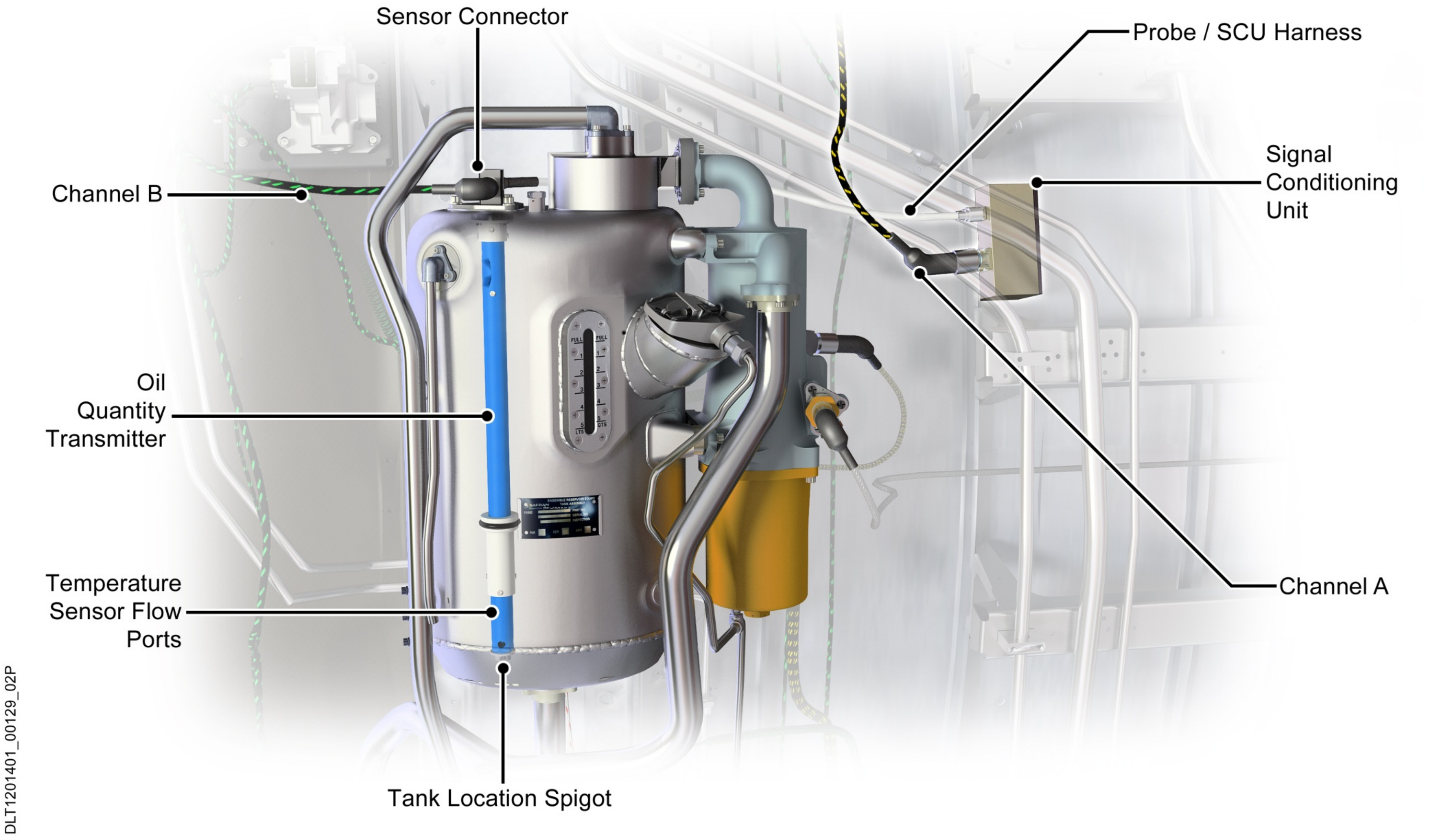
The sight glass and oil measurement scale are on the outer side of the oil tank.

Purpose

To give a visual indication of engine oil level for oil servicing.

Description

The sight glass lets the oil level in the oil tank to be visually checked, this indicates the quantity of oil to be added to replenish the system. There is a measurement scale either side of the sight glass, one part of the scale is identified in litres with the other part identified in US quarts. An anti-siphon tube adjacent to the sight glass supplies oil cleaning the sight glass during engine operation.



OIL QUANTITY TRANSMITTER AND TEMPRATURE SENSOR

Oil Scavenge Filter Differential Pressure Transducer

Location

The Oil Scavenge Filter Differential Pressure Transducer is mounted on the ODMS housing on the right hand of the oil tank assembly.

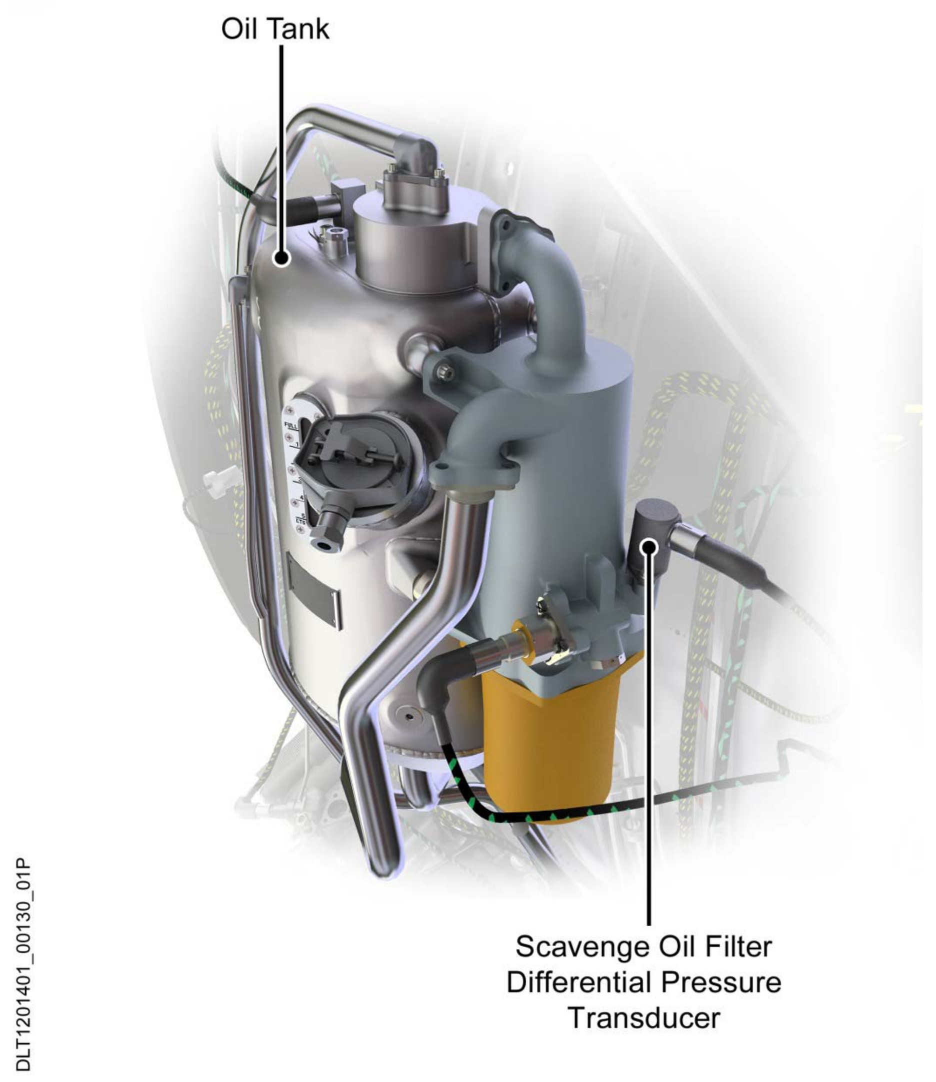
Purpose

To provide a cockpit indication of the filter blockage of the oil scavenge filter.

Description

Two pressure ports provide scavenge filter element inlet and outlet pressure to opposing sides of the sensing element within the transducer. Changes in the pressure balance across the sensing element cause a change in the resistivity of the element thus allowing direct measurement of the pressure drop between the two sides of the filter. A maintenance message is sent to the cockpit using channel B via the EEC to advise of impending oil pressure filter blockage, if the measured pressure difference is above the pre-defined threshold of 13 psid.

The scavenge filter by-pass valve opens when the measured pressure difference is above 20 psid; this enables maintenance activity in reaction to the impending blockage message to be taken prior to the valve being activated.



OIL SCAVENGE FILTER DIFFERENTIAL PRESSURE TRANSDUCER

Oil Pressure Transducer

Location

The Oil Pressure Transducer is mounted on the upper face of the FOHE housing.

Purpose

Provide engine oil pressure indication and low oil pressure warnings to the EEC and cockpit.

Description

The duplex output differential pressure transducer is mounted by means of a four bolt flange on to the Fuel Oil Heat Exchanger (FOHE) housing. The Transducer measures the pressure differential between the outlet from the FOHE and the internal gearbox (IGB) scavenge line. High pressure (FOHE outlet) is supplied to one side of the sensing element and low pressure (IGB scavenge) to the other side, a change in pressure differential cause a change in the resistivity of the sensing element thus allowing direct measurement of the pressure drop between HP and internal gearbox scavenge lines. The transducer is supplied with regulated voltage [10Vdc] by the EEC and provides two separate output signals one to each channel of the EEC. The EEC continually cross-checks the two signals, in the event of a cross-check failure the higher of the two transducer output values is used to provide the EEC output for cockpit display. The Oil Differential Pressure Transducer also provides low oil pressure alerts when the pressure falls below predetermined values.

The transducer is designed to measure the differential pressure range from -50 psid to 400 psid. The differential measurement is taken from the outlet of the FOHE and the internal gearbox scavenge line.

Low oil pressure switch

Location

The low oil pressure switch is mounted on the upper face of the FOHE housing.

Purpose

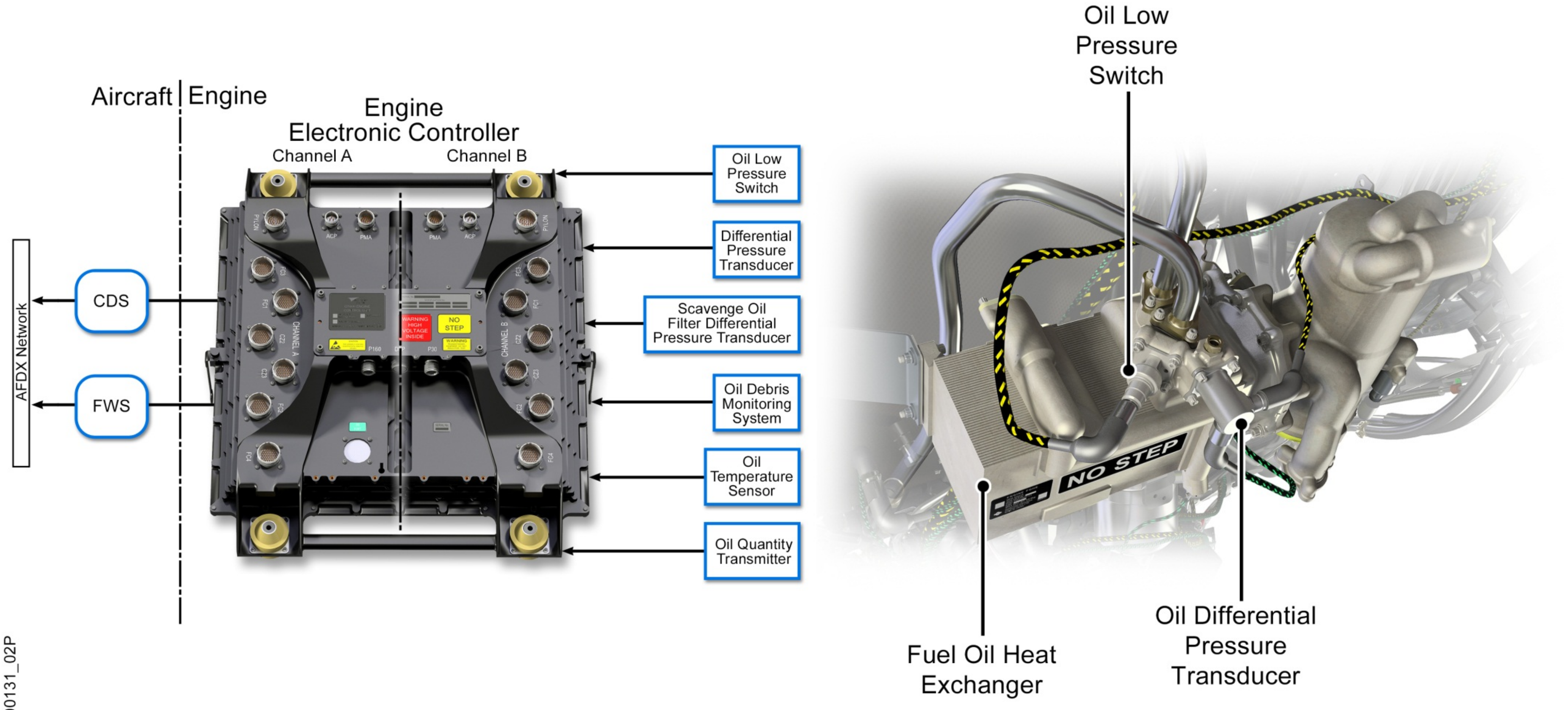
Provide an indication of low oil pressure through the engine to the EEC and cockpit.

Description

The switch provides a signal to Channel A of the EEC when the differential pressure between the outlet to the FOHE and the internal gearbox scavenge line increases over the predetermined level of 25 psid. The switch consists of a diaphragm which is exposed to FOHE outlet (High) and IGB scavenge (Low) pressures. The force developed by the differential pressure opposes a pre-set Belleville spring load and the switch actuates when force exceeds the pre-set Belleville load. When the differential pressure decreases to the predetermined level, the Belleville snaps to its original position.

Indication

When the oil pressure falls below the threshold value a Red Alert is set. This is derived from the 3 inputs, the oil pressure transducers and low oil pressure switch. With the engine running 2 out of the 3 inputs are required to enunciate the low oil pressure condition.



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OIL PRESSURE TRANSDUCER / LOW OIL PRESSURE SWITCH

FBH Vent Valve

Location

The FBHVV is on the right side of the LP compressor case approximately at the 4 o'clock position.

Purpose

The front bearing housing vent valve (FBHVV) controls the pressure in the vent circuit related to the Front Bearing Housing (FBH) compartment at changeable engine/flight conditions.

Description

The FBHVV has an inlet and an outlet for scavenge oil/air flow and an inlet and an outlet for actuation fuel flow. The unit is a two position valve commanded open or closed, as required, by the EEC to control the vent flow and maintain FBH chamber pressure and sealing.

a) Close when de-powered (solenoid de-energised).

b) Open when powered (solenoid energised).

In order for the control system to monitor the valve position feedback, FBHVV positional feedback is provided by a dual channel LVDT.

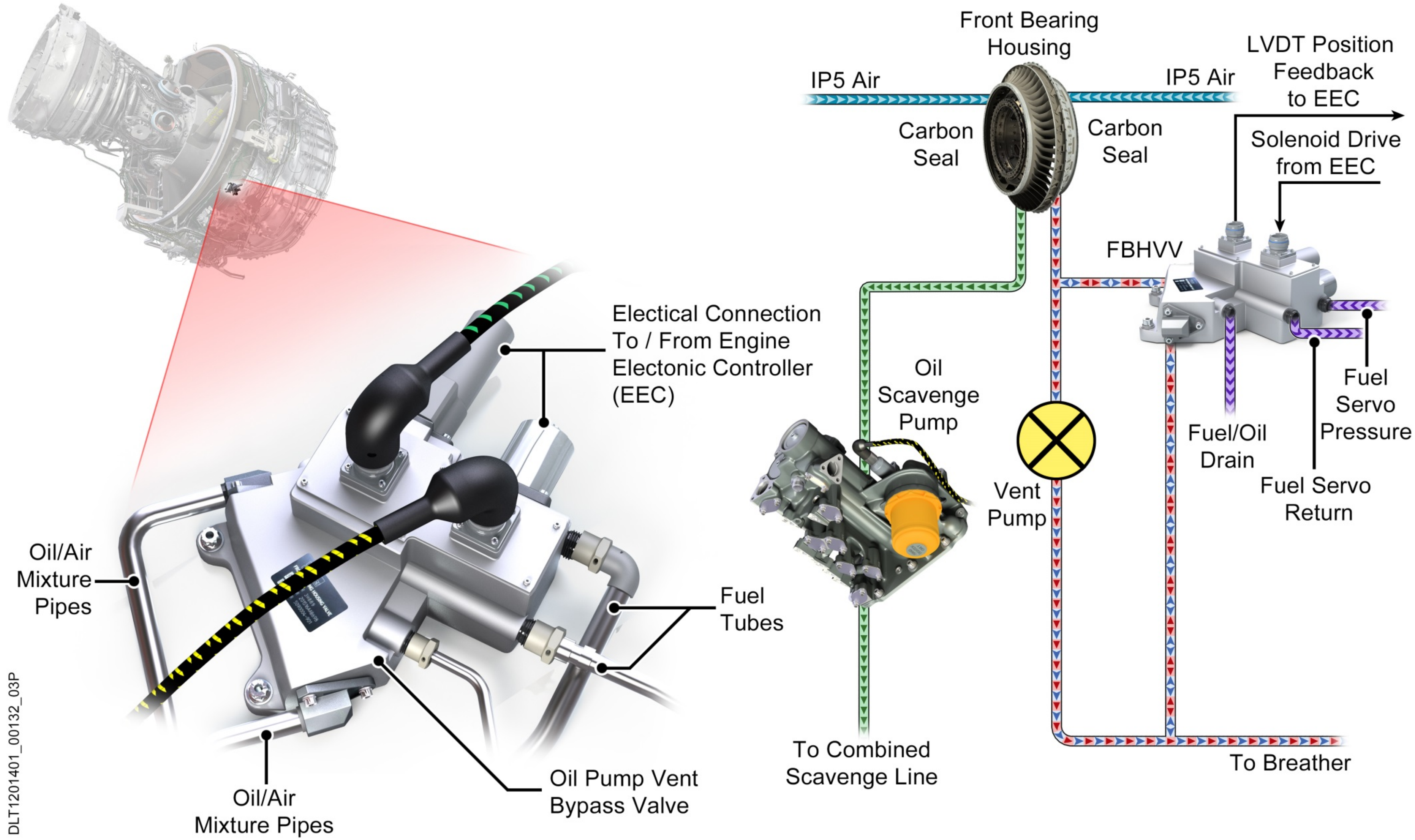
Operation

The FBHVV will be closed at low power conditions allowing air within the FBH to be removed by the FBH vent pump element. At these conditions, the breather pressure is higher than the IP5 pressure outside the FBH chamber, thus requiring the use of the vent pump to drive the vent flow.

At high power settings, the IP5 pressure will be higher than the breather pressure. At these conditions, the FBHVV will be open and FBH vent airflow will bypass the FBH vent pump in order to prevent the FBH vent pump element acting as a restrictor in the vent line.

At other conditions, such as flight idle at altitude, the FBHVV will be open and the airflow through the valve may reverse in order to prevent the FBH chamber pressure falling too low and leading to cavitation of the FBH vent and scavenge pump elements.

The FBHVV is designed such that it is not possible to install the FBHVV in an incorrect orientation.



FRONT BEARING HOUSING VENT VALVE

Ventilation System

Location

The ventilation system consists of two main units:

1. A de-aerator inside the oil tank.

2. A Centrifugal Breather mounted to the front face of the external gearbox.

Purpose

The purpose of the ventilation system is to separate the oil from the engine internal sealing / cooling air, so the oil can be reused and the air expelled to the atmosphere.

Description.

A controlled amount of air from various stages of the compressor system is allowed to continuously flow through the bearing chambers to seal / cool the chambers and prevent oil loss.

The air entering the chambers must then be removed and generally the scavenge pumps have the capacity to remove the air from most of the chambers. To provide a greater airflow through the bearing chambers a vent system is used.

As the airflows through the bearing chambers large quantities of an oil/air mist is produced.

Oil Tank De-aerator

The function of the oil tank de-aerator is to remove the air from the oil/air mist entering the oil tank. The de-aerator is a cyclone type separator allowing the oil to drop into the tank and the air to be vented to the centrifugal breather for a further separating process via the tank pressure relief valve at the top of the tank.

Centrifugal Breather

The centrifugal breather has a rotor that contains retimet segments and is driven by the external gearbox.

Aerated oil from the bearing chamber vent system and the oil tank is delivered to the centrifugal breather. The aerated oil mixture is passed through the retimet segments, where the mixture is separated. The oil is centrifuged out and scavenged back to the oil tank by a dedicated scavenge pump. The air passes through the retimet segments into the hollow rotor and is vented to atmosphere via the oil breather section of the combined drains mast.

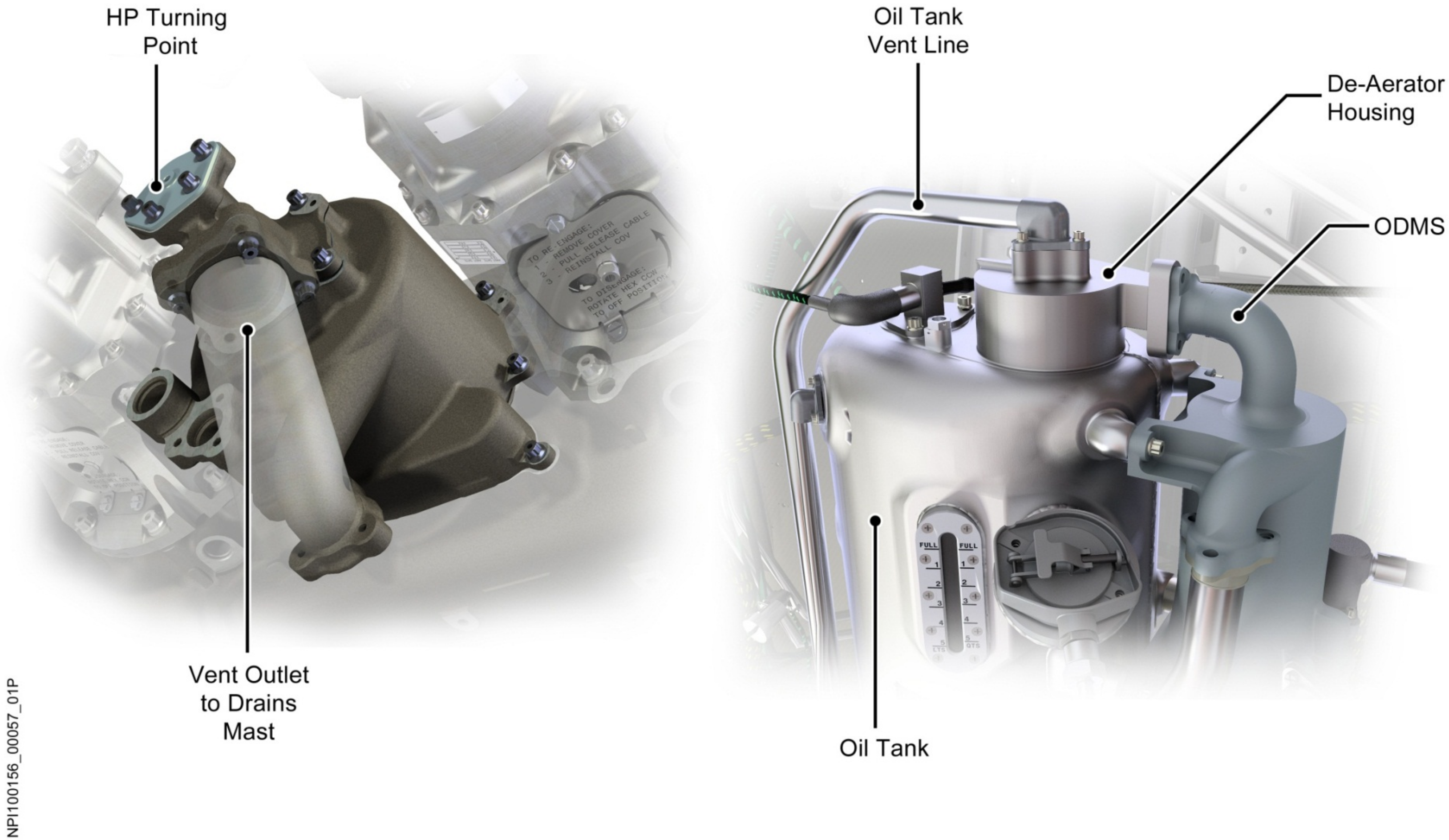
Modification

During low power operation a small amount of oil mist from the combined vent and oil tank vent is visible exiting from the breather exhaust mast.

A service bulletin details the installation of the hard ware to modify the system to include a coalescer unit which increases the oil mist particle size prior to entry into centrifugal breather. This reduces the visible breather mist at low power conditions.

Maintenance Note

The centrifugal breather houses the turning point for the HP system rotation.



VENTILATION SYSTEM COMPONENTS

Maintenance Practices

Oil Debris Sensor and Magnetic Chip Detector Inspection

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

MCD Inspection

TRENTXWB-A-80-11-00-03A01-280A-A

ODS Inspection

TRENTXWB-A-79-34-00-00A01-310A-A

1. Remove the ODS / MCD and remove the oil by cleaning using a clean container and kerosene.
2. Inspect the ODS / MCD using an inspection lamp and 10x magnifying lens. If debris is found on the ODS which is above the permitted limits install the MCD's to the oil pump assembly to identify the source of contamination and to help with fault isolation.

Note: MCD's are not installed for normal service as the debris catch on the ODS may be reduced. If the MCD's require fitment, install into their respective location on the oil pump assembly and run the engine IAW the TSM requirements.

3. Keep a record of any ODS or MCD inspections made, recording the Engine Serial Number, stating where the material was found and the date the material was collected.

Types of Debris

There are four (4) main types of debris that may be found. They are:

Swarf

Swarf is unwanted material that can stay in the engine when it is assembled. The swarf comes from machine operations when components are made. When this material is broken it can look the same as gear or steel seal material. Examine carefully to ensure it is different from other contamination.

Magnetic Fines

Magnetic fines are very small steel particles, which show as a black sludge on the ODS or MCD. Fines can be from gears, bearings or engine wear.

Note: Fines from gears are rougher than other types of fines.

Metallic Flakes

Metallic flakes usually come from ball bearings, roller bearings and gear teeth. They are irregular in shape and must be examined to find their origin.

Ball bearing and their track flakes are usually circular with radial cracks, if clean, will have one side brighter than other flakes and may have scratches that go across each other.

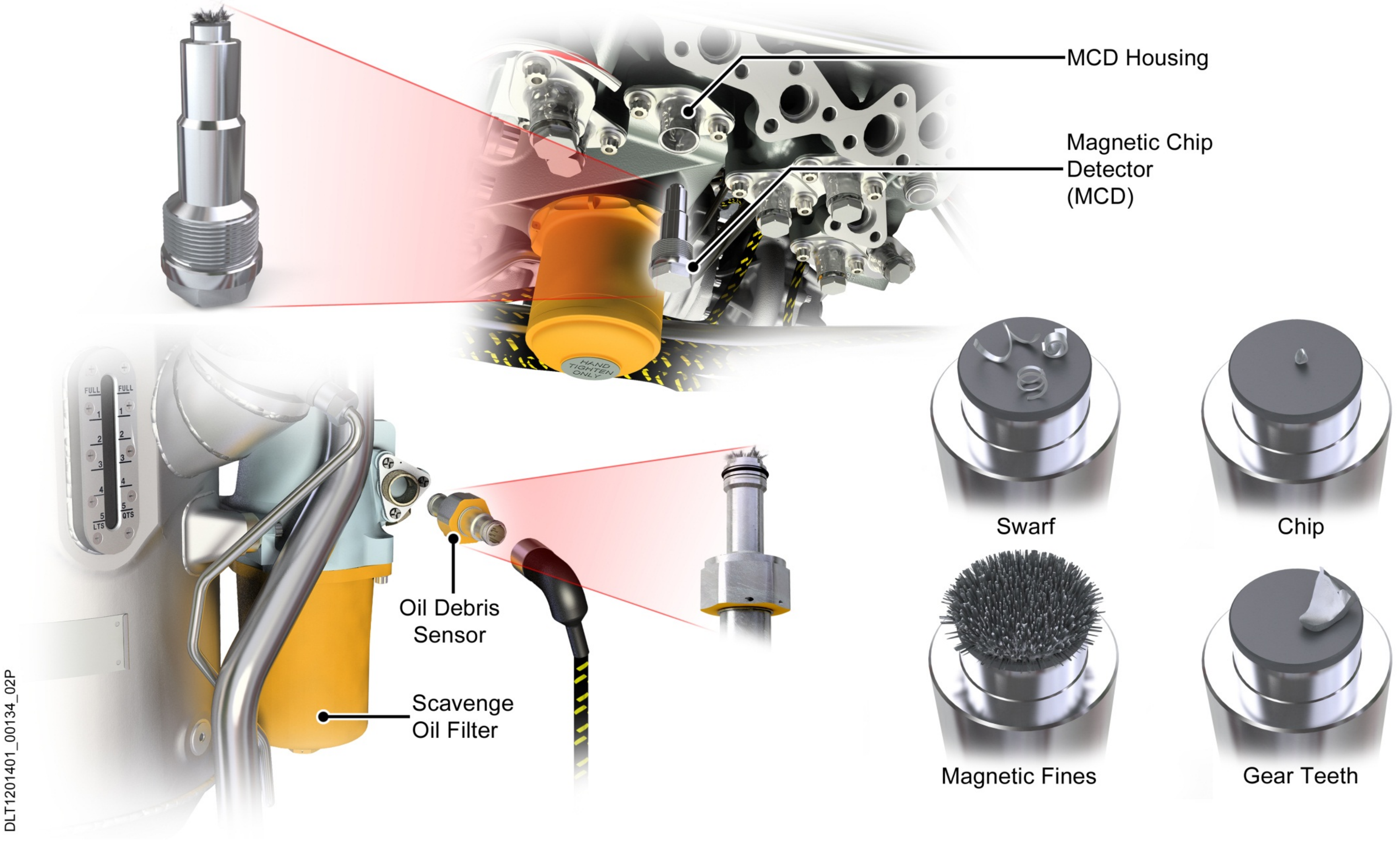
Roller bearing and their track flakes are usually rectangular. When clean will have one side brighter than other flakes and may have scratches that go across each other.

Gear teeth flakes are shiny, irregular in shape, thicker and not as bright as ball or roller bearing flakes.

Chips and Gear Tooth Fragments

Chips are very thick flakes or pieces of metal that usually have one smooth machined surface.

Fragments are corner pieces of gear teeth and may show that the gears are misaligned.



MCD AND ODS INSPECTION

Maintenance Practices.

Oil Scavenge Filter Removal / Installation

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Warnings and Cautions:

Observe all Warnings and Cautions given in the AMP

TRENTXWB-A-79-34-13-00A01-520A-A

Removal Procedure:

The procedure in the AMP is briefly described below:

- On the OMT, get access to the Power Distribution Control management pages and open & safety the applicable circuit breakers.
- Open the right fan cowl door.
- Put a clean 10 L container into position to catch the oil.
- Remove the drain plug from the filter housing and drain the oil into the container (Do not discard the oil at this step).
- Remove seal from the drain plug, install a new seal and refit the drain plug in the housing and torque.
- Hold the housing and remove the bolts and washers.
- Carefully remove the housing and filter from the scavenge filter cover.
- Examine the element and the drained oil for contamination (AMM 79-00-00-00A101-280A-A).
- Discard the element.
- Remove and discard the seal ring.

SCAVENGE OIL FILTER REMOVAL

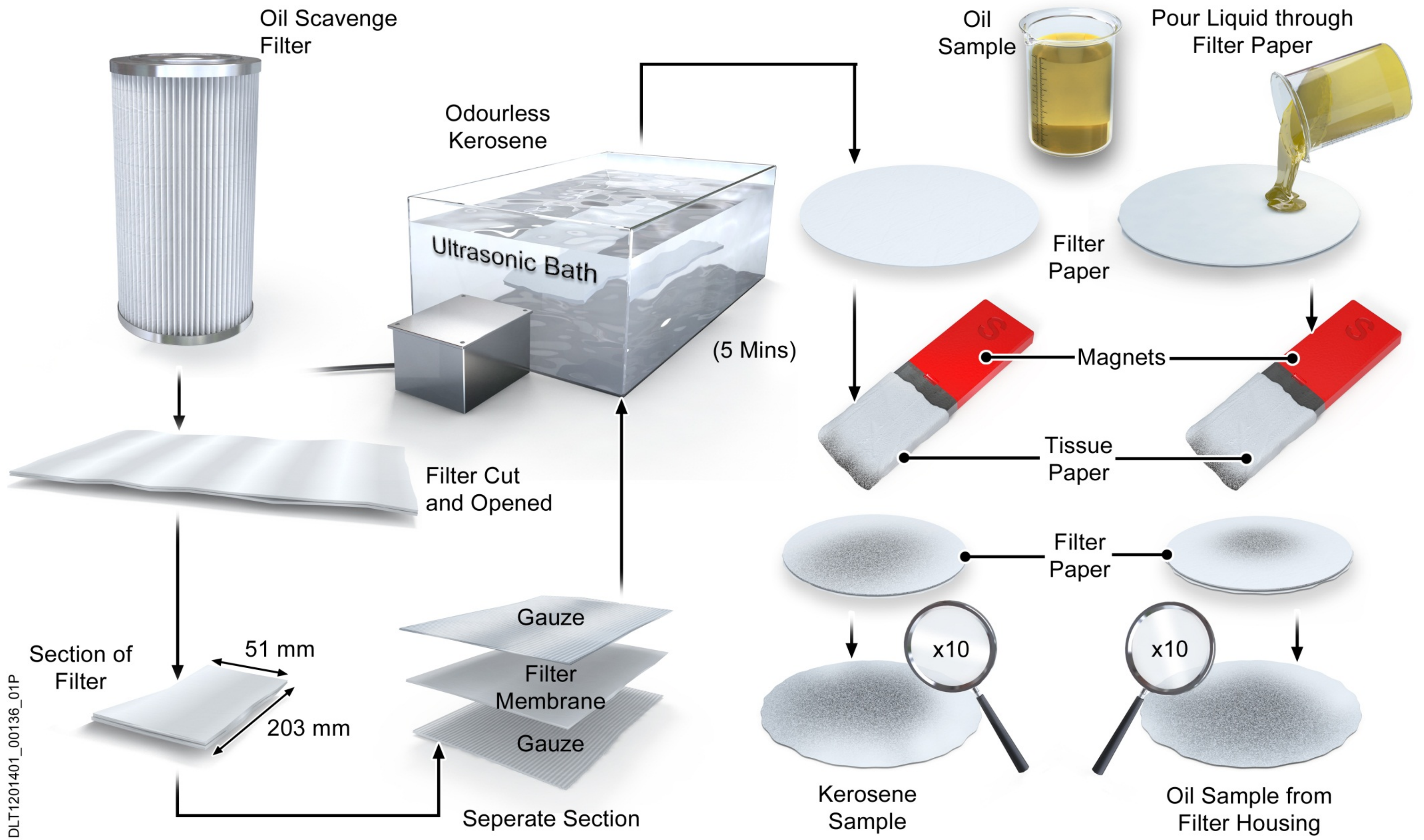
Maintenance Practices.

Oil Scavenge Filter Inspection

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.TRENTXWB-A-79-00-00-00A101-280A-A

Warning: BE CAREFUL WHEN YOU DO WORK ON THE ENGINE PARTS AFTER THE ENGINE IS STOPPED. THE ENGINE PARTS CAN STAY HOT FOR ALMOST ONE HOUR. DO NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE INJURY TO PERSONNEL.

1. Remove the contamination from the filter element.
2. Clean the filter and oil scavenge filter housing in a fuel/oil resistant, non-metallic beaker containing clean kerosene.
3. Lightly hit the filter element to remove any contamination.
4. Keep the used kerosene and contamination in the beaker for examination.
5. Remove the contamination from the filter membrane.
6. Cut the end caps off the element and make a cut longitudinally from one end to the other.
7. Discard the end caps.
8. Place the filter element on a flat surface and fully extend the element to make a flat sheet.
9. Remove a piece of the element approximately 51mm by 203mm (2 inches x 8 inches) to make a sample.
10. Remove the inner and outer gauze from the filter membrane. Do not contaminate the centre area with metal from the wire mesh.
11. Place the sample into the beaker with kerosene in and place into the ultrasonic bath.
12. Operate the ultrasonic bath for 5 minutes.
13. Remove the sample from the beaker. Do not discard the kerosene as it may contain debris that will help with your diagnosis.
14. Put the drained oil from the filter housing through a clean filter paper.
15. Place a tissue around a magnet and pass the magnet over the filter paper.
16. Put the kerosene from the beaker through a clean filter paper.
17. Place a tissue around a magnet and pass the magnet over the filter paper
18. Examine both tissue papers using a 10x-magnifying lens.
19. Refer to the AMP for the acceptance and reject standards. Laboratory analysis is also recommended to help with material identification.



OIL SCAVENGE FILTER INSPECTION

Maintenance Practices

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Warnings and Cautions:

Observe all Warnings and Cautions given in the AMP

TRENTXWB-A-79-34-13-00A01-720A-D

Scavenge Oil Filter Installation

Installation Procedure

- Examine the inner area of the housing and make sure it is clean and clear of unwanted material.
- Install a new seal ring to the housing.
- Carefully install the filter element in the scavenge filter cover. Make sure you hold the filter element in this position.
- Put the housing in position on the scavenge filter cover.
- Attach the housing with the bolts and washers.
- Torque the bolts to the value given in the AMP.
- Fill the engine oil system.
- Do a fuel and oil leak check of the scavenge filter housing.
- Put the aircraft back to its initial configuration.

OIL SCAVENGE FILTER INSTALATION

Maintenance Practices.

Pressure Oil Filter Removal

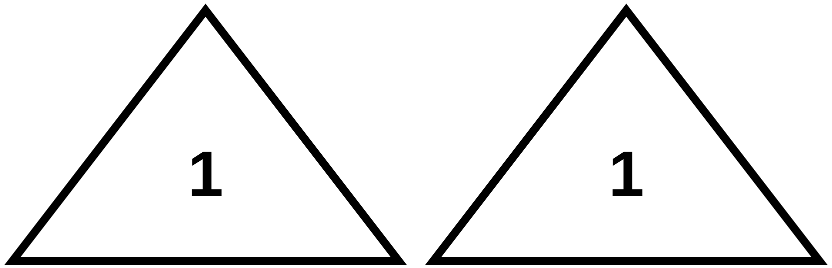
The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

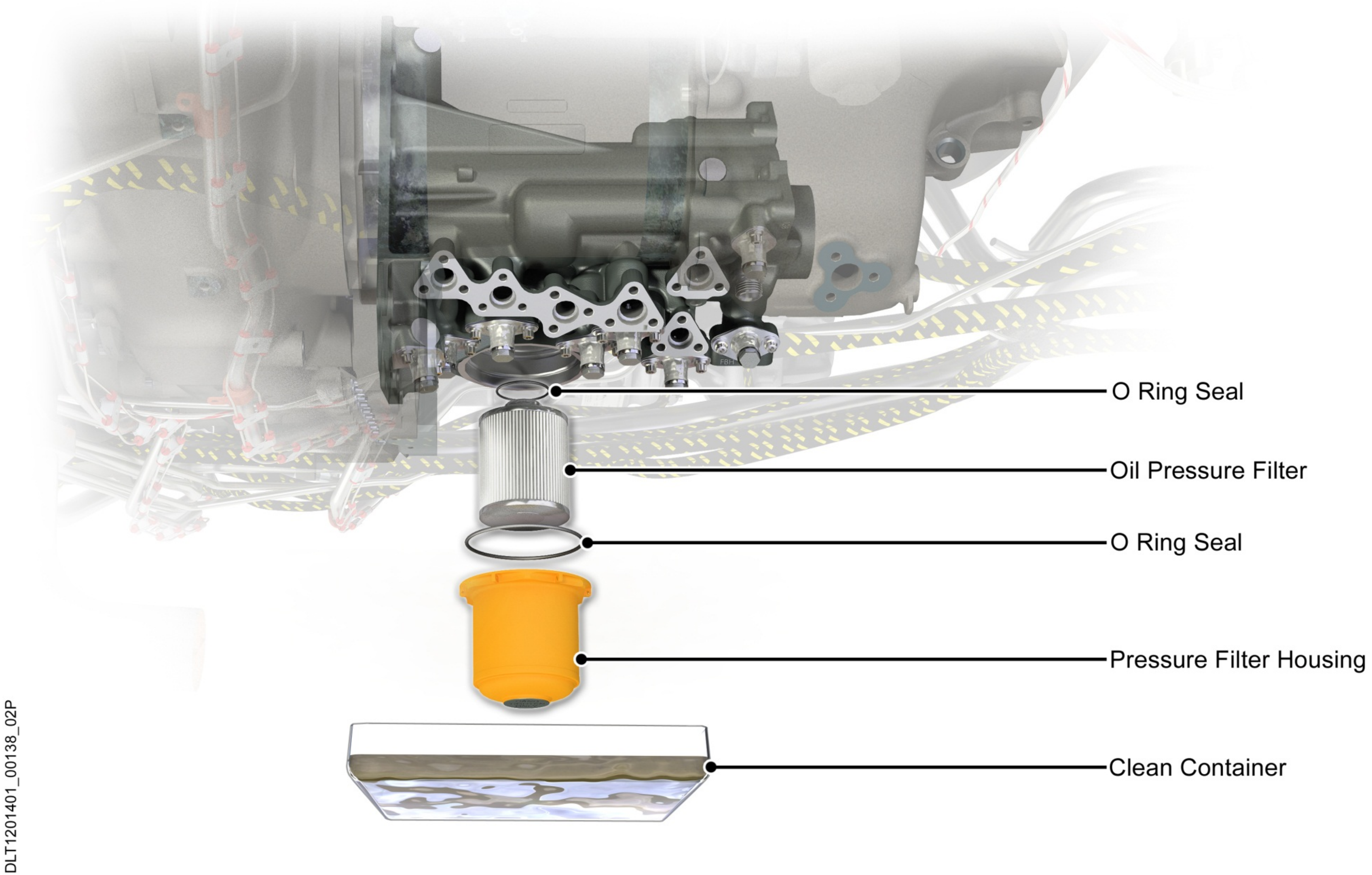
Oil Pressure Filter Removal

AMP TRENTXWB-A-79-22-11-02A01-520A-D

The Warnings and Cautions are the same as the Scavenge Oil Filter Removal and Installation.

1. Ensure the EEC MAINT switch is in the NORM position by accessing the maintenance Control Panel Systems Menu and selecting MAINT, MAINT CTRL PGS, and MISC SYSTEM CTRLS (MSC).
2. Place a 5-gallon (19 litres) oil resistant container below the oil pressure filter housing.
3. Remove and disregard the safety cable from the oil pressure filter housing.
4. Loosen the pressure filter housing by turning the housing counter clockwise. A strap wrench may be required for this task.
5. Continue to turn the oil pressure filter housing counter clockwise and remove the oil pressure filter and housing from the oil pump assembly.
6. Let the oil drain from the oil pressure filter housing into the 5-gallon container.
7. Pour the oil from the pressure filter housing into the clean container.
8. Remove the filter from the housing and discard the seal rings from the filter element and the housing.
9. Keep the oil pressure filter element and the oil from the container if a contamination check is required.
10. Inspect the filter (AMP **79-00-00-00A101-280A-A**).
11. If the filter is satisfactory then it can be cleaned a maximum of two (2) times.
12. After cleaning the filter should be engraved with a triangle and with a number one (1) inside the triangle.





OIL PRESSURE FILTER REMOVAL AND INSTALLATION

Maintenance Practices.

Oil Pressure Filter Inspection

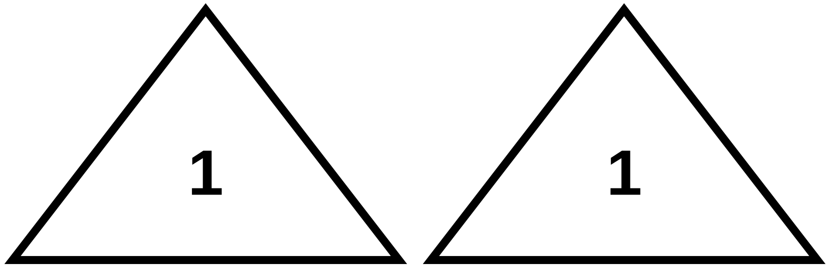
The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

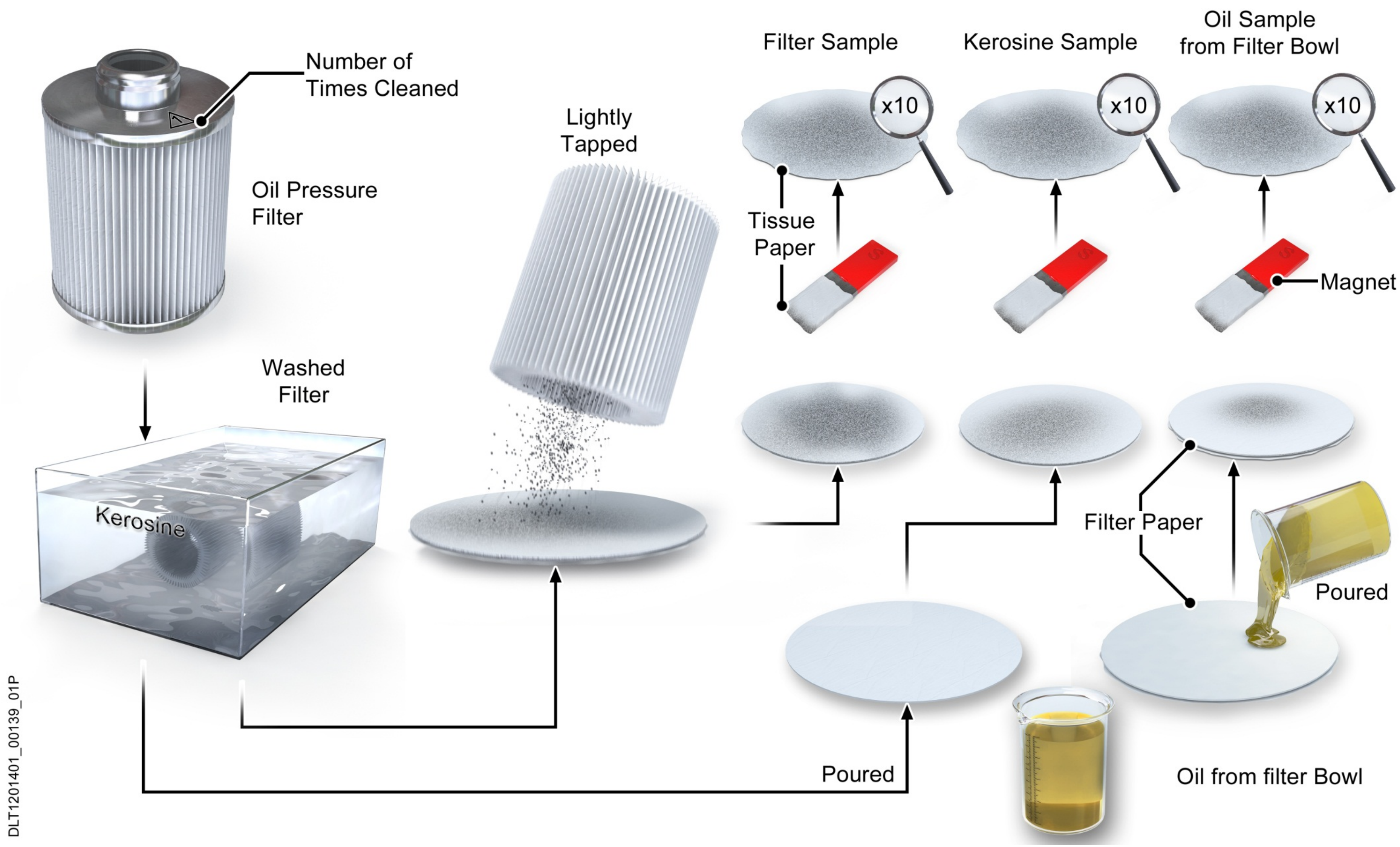
Pressure Oil Filter Inspection

AMM TRENTXWB-A-79-00-00-00A101-280A-A

The Warnings and Cautions are the same as the Scavenge Oil Filter Removal and Installation.

1. Examine the filter element, if damaged discard the filter.
2. If not damaged check how many times the filter has been cleaned. If the filter has two (2) triangular marks with a number inside then replace the filter.
3. If the filter has one (1) triangular mark with a number inside then clean the filter.
4. Temporarily blank off the ends of the filter and place into a clean solvent container.
5. Using clean solvent (refer to the AMP for the types of solvent) allow the filter to soak for between 30 minutes and 2 hours in a 19 litre (5 gal) container.
6. Place the filter in a 1 litre (1 Quart) container with clean solvent inside an ultrasonic cleaner.
7. Allow the ultrasonic cleaner to operate for 5 minutes.
8. Repeat operations 6 and 7.
9. Shake the filter and dry using clean compressed air.
10. Examine the filter using a 10x-magnifying lens.
11. If contamination material is evident discard the filter.
12. If there is no contaminations engrave a triangle with a number one inside to indicate the filter element has been cleaned.
13. If the filter is not to be installed immediately the place the filter in a plastic bag write the date on the bag and place into storage.
14. Refer to the AMP for the acceptance and reject standards. Laboratory analysis is also recommended to help with material identification.





OIL PRESSURE FILTER INSPECTION

Maintenance Practices.

Oil Pressure Filter Installation

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Pressure Oil Filter Installation

AMM TRENTXWB-A-79-22-11-02A01-720A-D

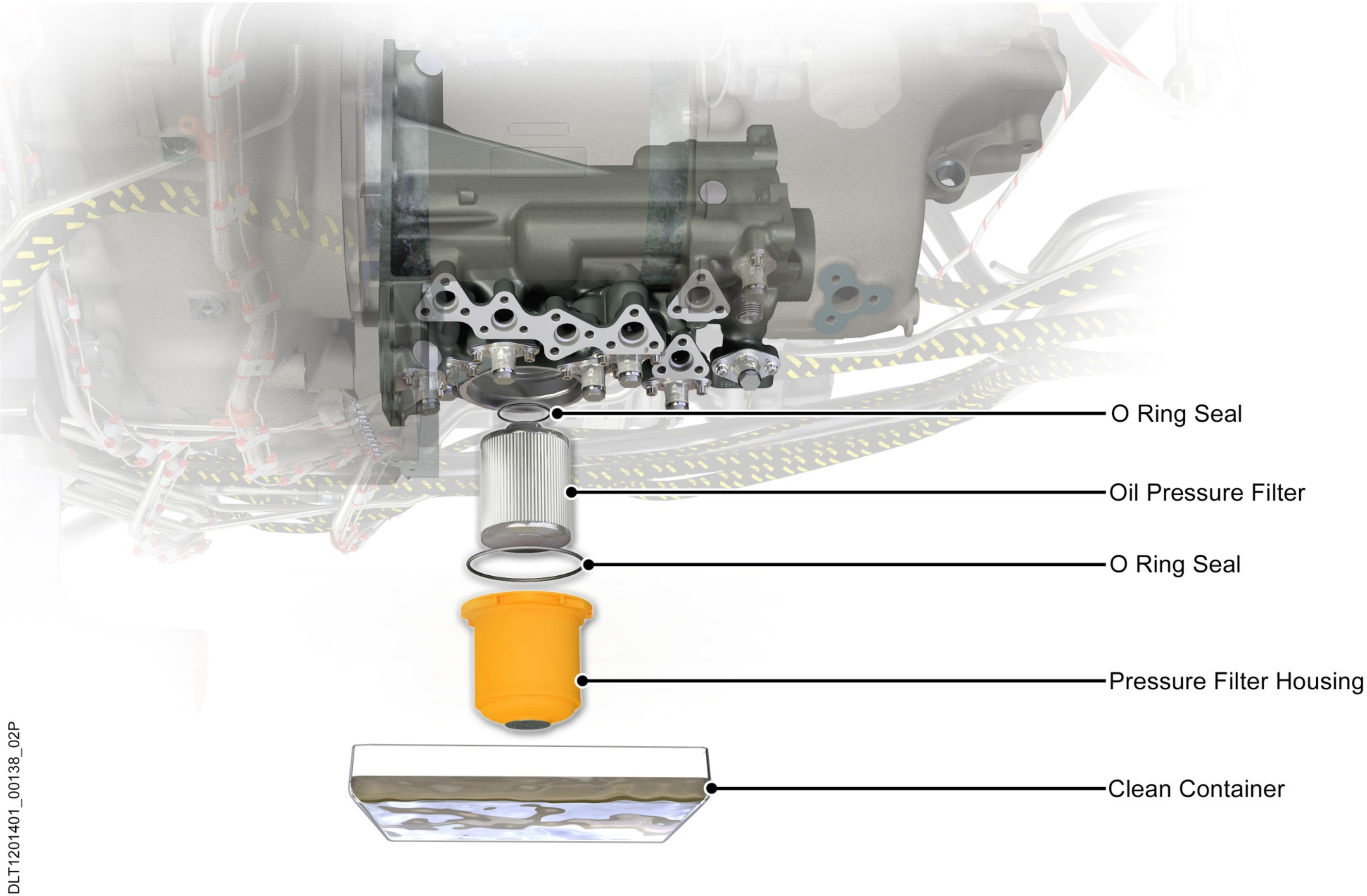
The Warnings and Cautions are the same as the Scavenge Oil Filter Removal and Installation.

1. Install new seal rings to the oil pressure filter and housing (TRENTXWB-A-70-02-01-00A01-950A-A).
2. Install the filter element ensuring the filter engages into the oil pump assembly.
3. Install the oil pressure filter housing into the pump ensuring that the threads of the housing engage correctly.
4. Turn the filter housing clockwise by hand until tight.

CAUTION

TIGHTEN THE FILTER HOUSING WITH YOUR HAND ONLY, IF YOU USE TOOLS, DAMAGE CAN OCCUR

5. Once tight safety the filter housing to the oil pump assembly using the recommended safety cable.
6. Replenish the engine oil as required.
7. Carry out the applicable test for the filter element (TRENTXWB-A-71-00-00-00A01-320D-A).



OIL PRESSURE FILTER INSTALLATION

Maintenance Practices.
Oil Servicing

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

If the engine has been stopped for not more than 6 hours use the oil sight glass to do a visual Inspection of the engine oil level.

If the engine has been stopped for more than 6 hours do the steps that follow:

Start the engine; refer to TRENTXWB-A-71-00-00-00A01-950A-A.

Operate the engine at idle for a minimum of 10 min.

Stop the engine; refer to TRENTXWB-A-71-00-00-00A01-950C-A.

Procedure

Use the oil sight glass to do a visual inspection of the engine oil level.

You must wait for a minimum of 10 min after the engine has stopped before you do a check of the oil level. This will let the oil level become stable.

If the oil level is below the FULL mark of the oil tank sight glass and you can see oil in the oil sight glass, fill the Oil tank.

If the engine has been stopped for between 2 h and 6 h, the oil level in the Oil tank can be 1 US quart below the Oil tank full mark. No further action is required.

Fill the Oil tank

Open the Cap

Pull the lever on the Cap to the unlock position.

Turn the lever on the Cap counter clockwise to the open position and open the Cap

If you can smell fuel in the Oil tank when you remove the Cap do the step that follows.

Do a check for fuel fumes in the oil tank; refer to TRENTXWB-A-79-00-00-00A01-390C-A.

Note

Only do this step if you think that you can smell fuel fumes in the oil tank.

Note

Fuel fumes are easier to find when the contents of the oil tank are hot.

Pour clean Engine oil into the Oil tank

Visually examine the Seal ring. Make sure it is fully engaged in the groove of the Cap and is not loose, missing or damaged.

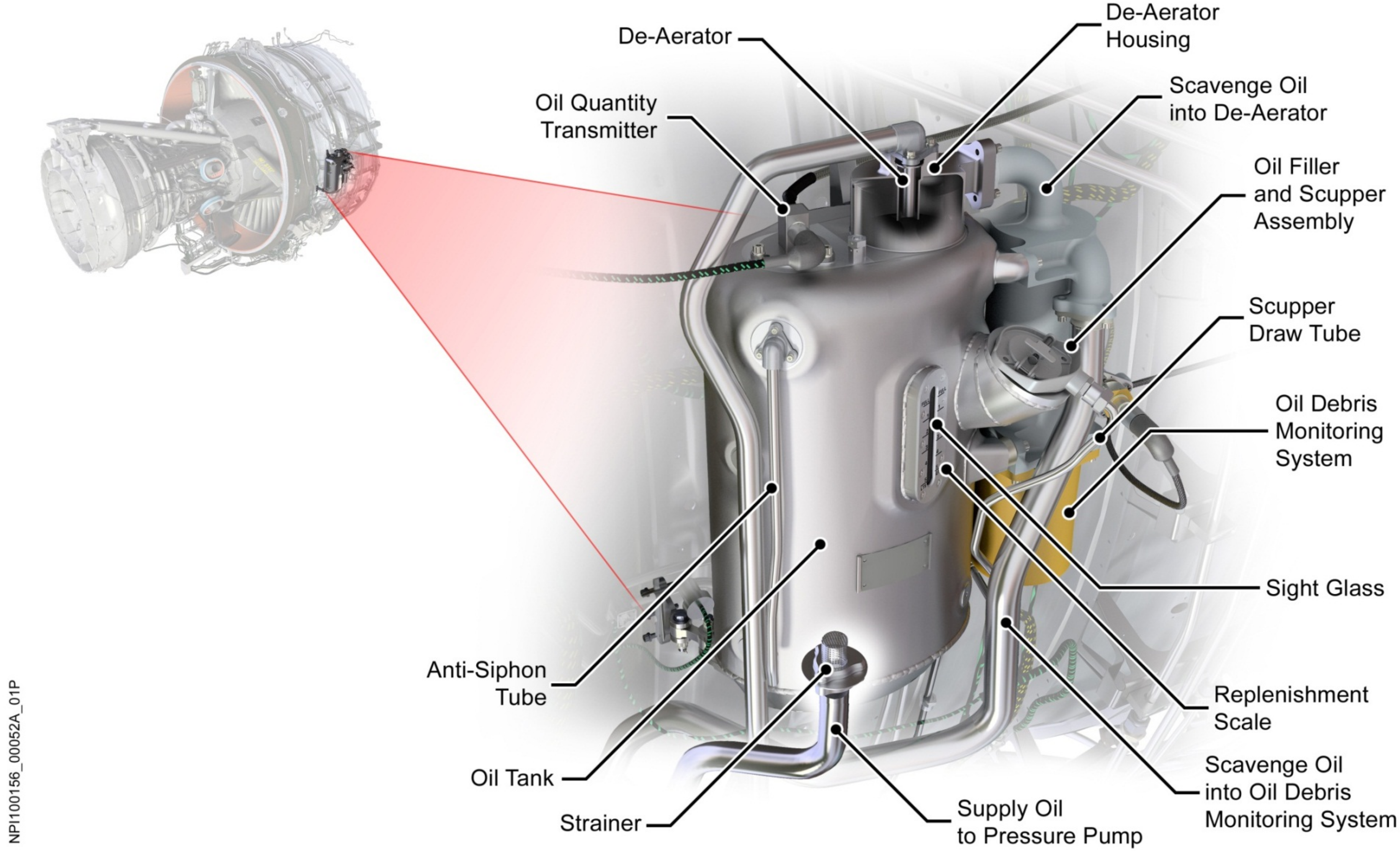
If the Seal ring is damaged replace the Seal ring, refer to TRENTXWB-A-70-02-01-00A01-950A-A.

Close the Cap

Turn the lever on the Cap clockwise to the closed position.

Push the lever down to the locked position.

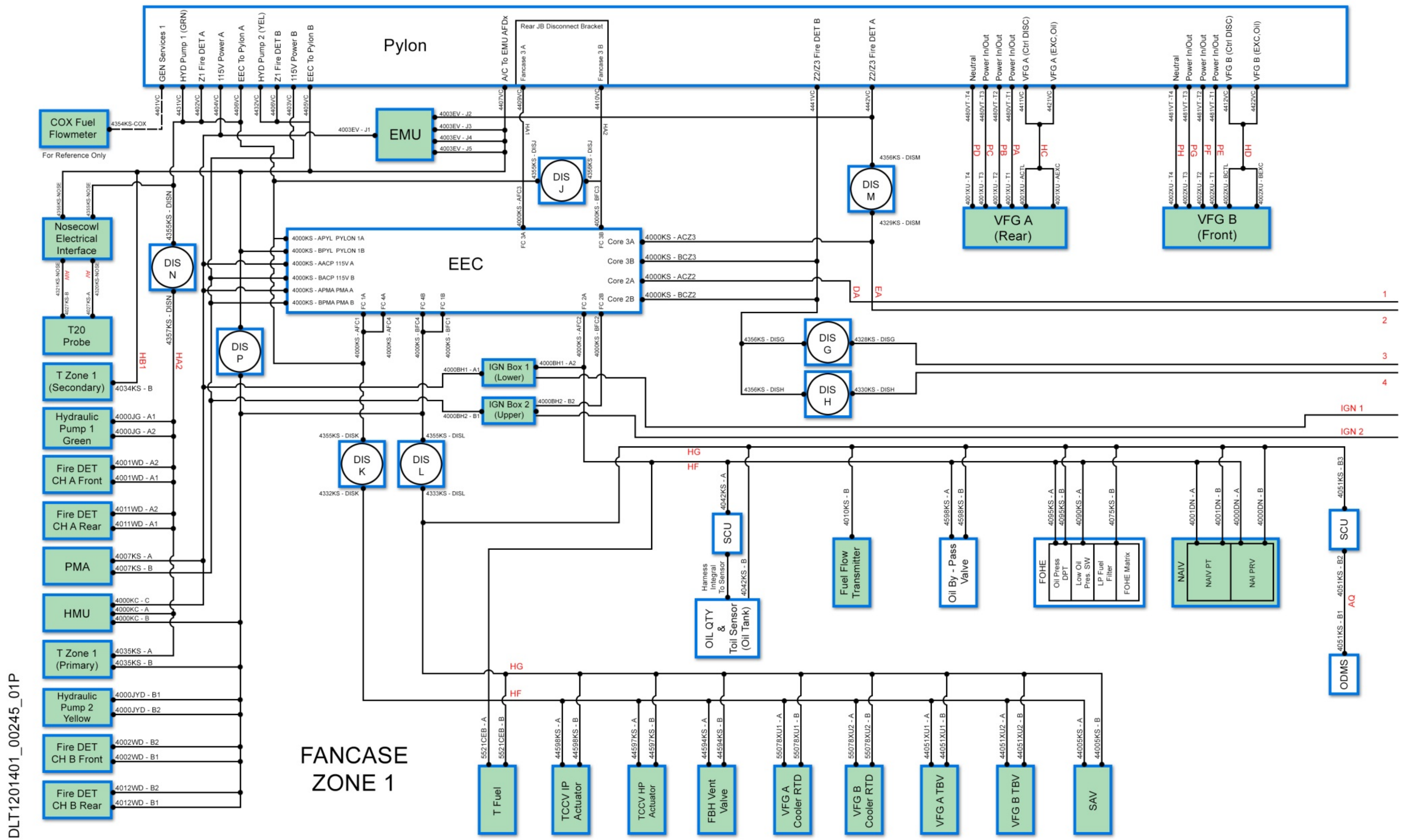
Record the quantity of Engine oil added.



OIL SERVICING

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.



ZONE 1WIRING DIAGRAM

Section 7 – Engine Oil System

Objectives

At the end of this section the student should be able to:

- State the purpose of the Engine Oil lubrication System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Oil Lubrication System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the engine Oil Lubrication System of the Trent XWB engine.
- State the WARNINGS & CAUTIONS associated with the engine Oil Lubrication system of the Trent XWB engine.
- Describe how the Trent XWB engine Oil Lubrication System interfaces with other engine and aircraft systems.

End of Oil Section

Section 8 - Engine Fuel and Control System

Section 8 - Engine Fuel and Control System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Fuel and Control System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Fuel and Control System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Engine Fuel and Control System of the Trent XWB engine.
- State the WARNINGS & CAUTIONS associated with the Engine Fuel and Control System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Fuel and Control System interface with other engine and aircraft systems.

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Power Control System (PCS)

Thrust Control and Management

Location

The software and hardware that controls engine thrust is located on both the engine and within the aircraft.

Purpose

The purpose of thrust control is to ensure the engine provides the correct level of thrust for safe aircraft operation at all conditions.

Description

The components used to control thrust are:

- A thrust lever (one per engine)
- A reverse lever (one per engine)
- The EEC
- Engine sensors
- Hydro-mechanical unit (HMU)
- Primary Flight Control Computer (PRIM)
- Throttle Control Assembly (TCA)

The thrust of each engine can be manually or automatically controlled. Manual control of forward thrust is by the use of levers on the flight deck and automatic control is by the use of software functions within the PRIM.

There is one combined lever in the cockpit for each engine, one function is for forward thrust (throttle lever) and the other for reverse thrust (reverse lever).

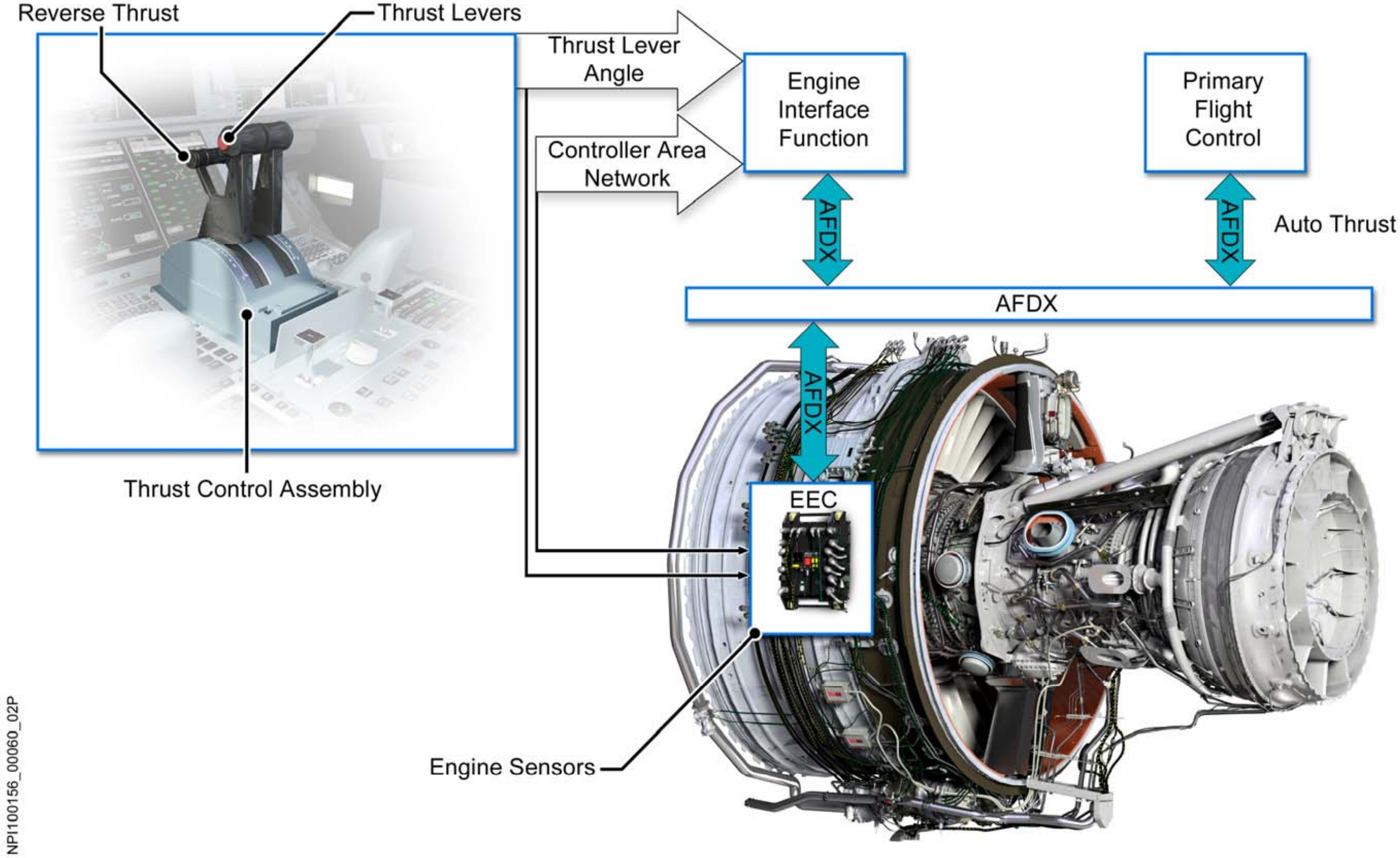
The throttle lever controls the forward thrust of the engine by interfacing with the EEC using electrical cables via the AFDX system. The EEC uses these signals and the engine sensors to calculate the required fuel flow for the demanded thrust and interfaces with the HMU through electrical cables.

The reverse lever operates the opening and closing of the thrust reverser translating sleeves. The reverse lever also allows the flight crew to change the amount of thrust required to slow the aircraft during breaking on landing.

The engine control system uses LP corrected shaft speed (NL) for primary thrust setting and control. The NL thrust command is derived from thrust ratings that define an NL to thrust relationship for a particular throttle position. The ratings include Maximum Take-off (MTO), Maximum Continuous (MCT), Maximum Climb (MCL), Forward Idle, Reverse Idle and Maximum Reverse Thrust. Provision is also made for FLEX, and pre-set de-rates.

Multiple engine ratings are available for use in primary thrust mode defined as Trent XWB-75, Trent XWB-79, Trent XWB-79B and Trent XWB-84. These allow the correct thrusts to be selected for different aircraft variants. Each of these engine ratings consists of a subset of thrust ratings (i.e. MTO etc.). The engine rating is selected via the Data Entry Plug (DEP).

The forward thrust ratings are designed to have adequate sensitivity and accuracy by taking into account air data parameters (ambient temperature, ambient pressure, Mach number and delta temperatures and pressures) and cabin bleed air and anti-ice bleed air demands from the aircraft. The bleed air demands are frozen during take-off to prevent sudden thrust changes.



ENGINE THRUST CONTROL

Auto thrust (A/THR) Configuration

The Auto thrust (A/THR):

Location

The auto-thrust system components are located on both the engine and the aircraft.

Purpose

- Controls the thrust of the two engines via control orders to Full Authority Digital Engine Control (FADEC)
- Holds selected or managed speed / Mach
- Holds thrust and performs thrust reduction during flare
- Ensures protection against high angle of attack, (Alpha Floor Protection)

Operation

The A/THR can operate independently or with the Auto Pilot/Flight Director (AP / FD):

- If the AP / FDs are off, the A/THR still controls the speed or Mach
- If AP and / or FDs are engaged, the A/THR mode and the AP / FD vertical mode are linked

The A/THR can be active in all phases of flight except during take-off phase.

Selection

A/THR Arming / Activation / Disconnection
The flight crew can arm the A/THR via the A/THR push button on:

- Auto Flight System (AFS) Control Panel (CP)
- Flight Control Unit (FCU) AFS backup page on the Multi-Function Display (MFD)

The A/THR is armed automatically at take-off or go-around and active when thrust levers are in the correct detent (CLB or MCT) whereas in alpha floor, A/THR is directly active whatever the detent position.

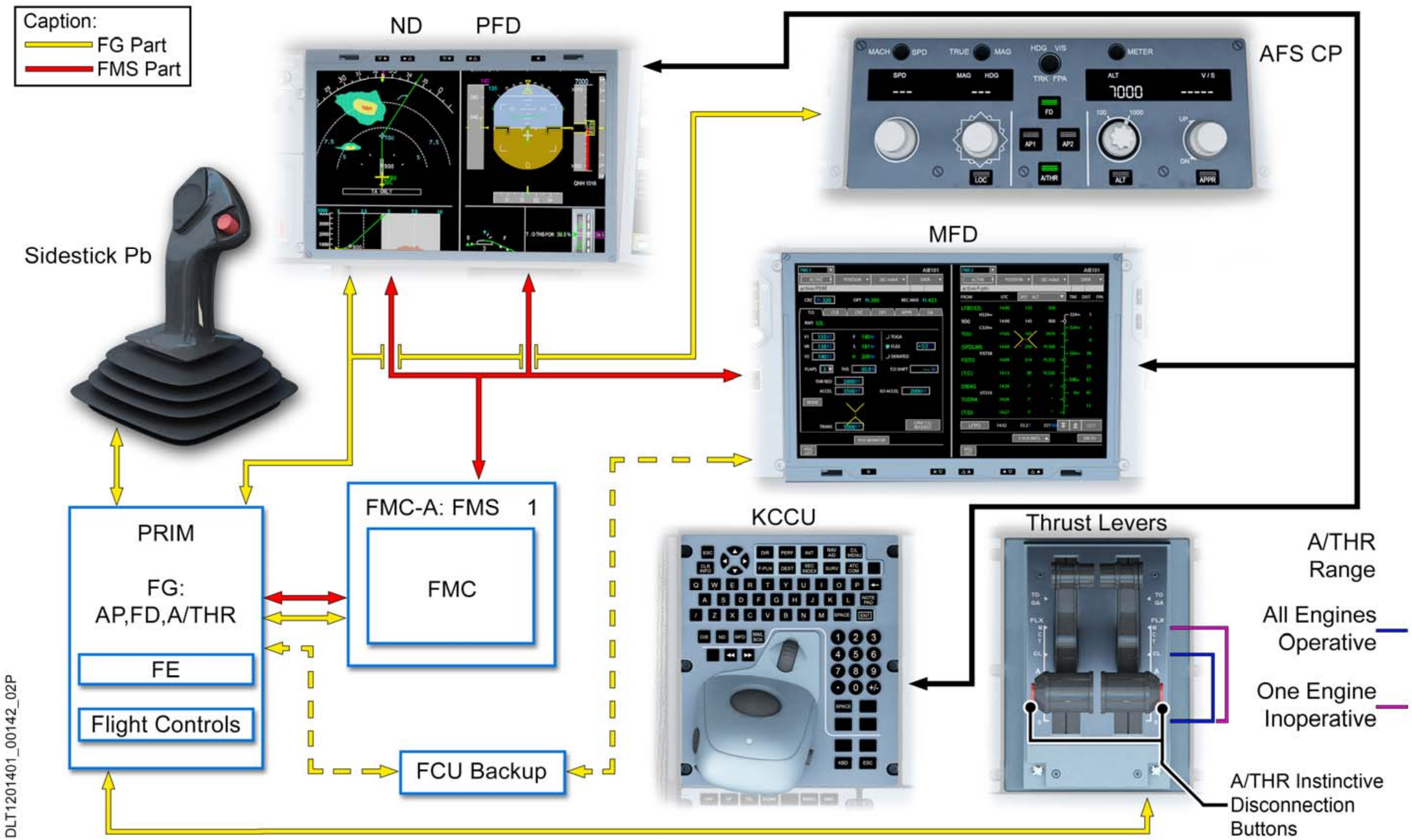
The flight crew can disconnect the A/THR via the:

- Instinctive disconnection button
- A/THR push button
- On the AFS control panel
- On the FCU AFS backup page on the MFD
- Thrust levers, if the flight crew sets the two levers to idle

Avionic systems interfaces

Other aircraft avionic systems which interface with the Auto Thrust system are:

- Primary Flight Display (PFD)
- Navigation Display (ND)
- Primary Computers (PRIM)
- Keyboard & Cursor Control Unit (KCCU)
- Flight Management Computer (FMC)
- Flight Envelope parameters (FE)
- Flight Control Unit (FCU)
- Flight Guidance (FG)



COCKPIT AUTO THRUST CONFIGURATION

Thrust Modes

Location

The thrust modes controlling the engine are located within EEC software and the Propulsion Control System (PCS).

Purpose

To provide different modes to which the flight crew can control the thrust of the engine for different conditions.

Description

The different modes of thrust control are as follows and are indicated on the ECAM display in the cockpit:

Manual Mode

The throttle lever is moved by the flight crew to select the required thrust. The EEC converts the throttle position signal to an N1 corrected speed. There are four forward detent positions:

- Forward Idle (0)
- Climb (CL)
- Flex Take-Off (FLX) / Max Continuous (MCT)
- Max Take-Off and Go-Around (TOGA)

And a variable position piggy back thrust lever that operates between two positions:

- Reverse Idle
- Reverse Max

Automatic Thrust Control (Auto thrust)

The EEC, from the PRIM, receives a target thrust signal that is converted by the EEC to an N1 corrected speed. Auto thrust is selected when the throttle lever is moved to the Climb (CL) detent position and disconnected by a button on the throttle lever. The throttle lever will not move in auto thrust.

Forward Thrust

In manual and auto thrust modes the EEC, in response to inputs from the throttle lever, controls engine forward thrust. This is called N1 Primary Thrust.

Reverse Thrust

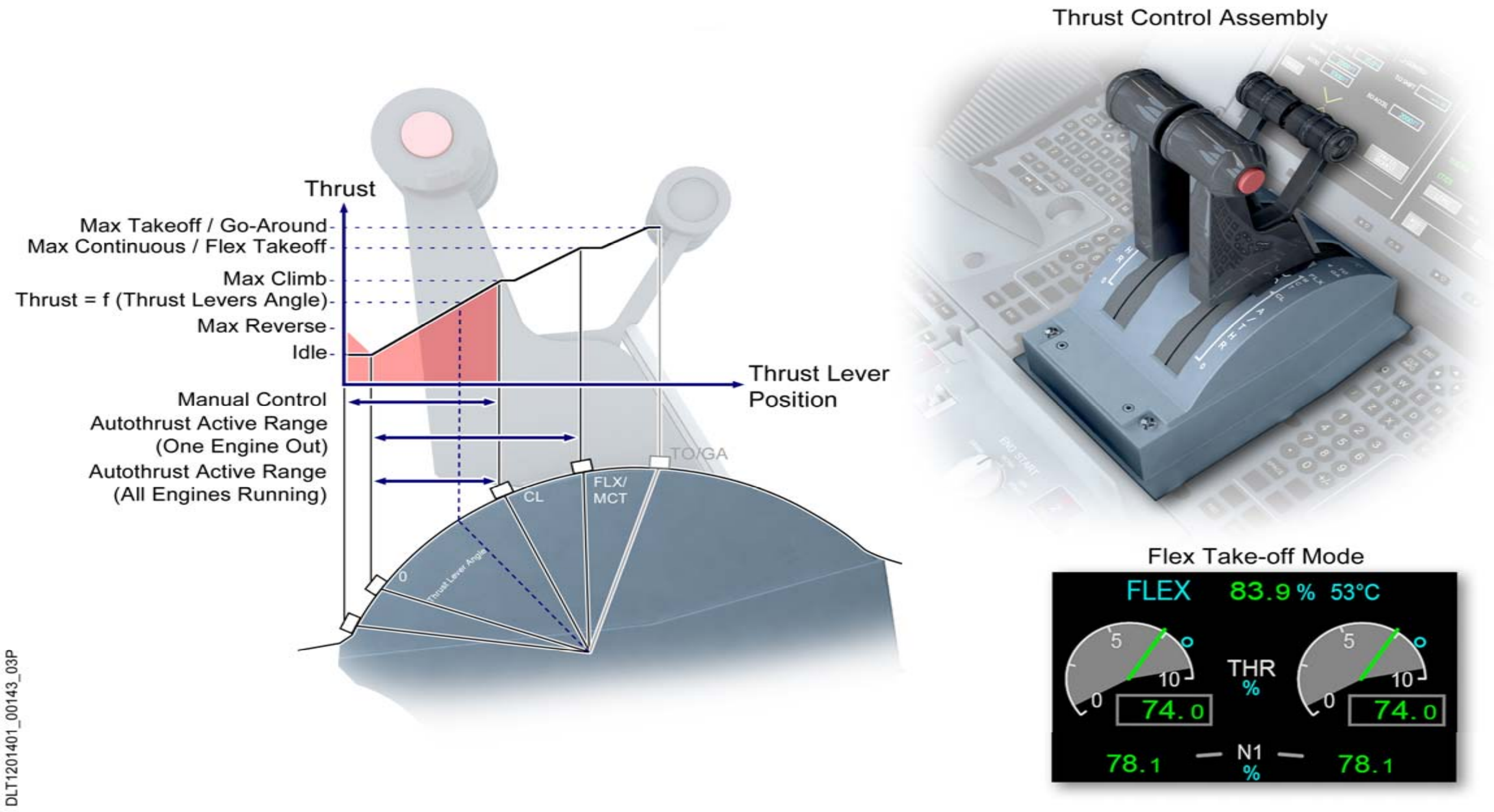
In reverse thrust the EEC interfaces with the thrust reverser unit when the throttle lever is in the idle position and the reverse lever is lifted. The EEC controls the engine to an N1 speed that can be modulated between Reverse Idle and Maximum Reverse by the flight crew.

Primary (THR) Thrust Control

The primary control parameter of the Trent XWB is N1 Primary Thrust and is indicated in the cockpit as a percentage of thrust. The Engine Control System uses air data parameters to continuously derive the most accurate N1 speed thrust level possible for primary control of engine forward / reverse thrust. This is a change from earlier Trent engines, which use either EPR (Engine Pressure Ratio) or TPR (Turbofan Power Ratio). LP speed was chosen for the Trent XWB because it is a simple, robust measurement suited to high bypass ratio engines where the fan and core systems may be split for transportation.

N1 Unrated Thrust

The EEC will automatically change to N1 Unrated Thrust when N1 Primary Thrust cannot be calculated.



ENGINE THRUST SELECTION

Thrust modes continued

Flex Take-Off / Fixed Derate Settings

Flex take-off / Fixed Derate modes allow the take-off to be performed at a decreased level of thrust when the aircraft payload is light and conditions allow. De-rated settings are selected via the Keyboard and Cursor Control Unit (KCCU) in conjunction with the Multi-Function Display (MFD) and Flight Management System (FMS).

Flex take-off is selected by the Flight crew and is based on inputting fictitiously high Outside Air Temperatures (OAT) from algorithmic data, based on aircraft all-up weight, airfield parameters and atmospheric conditions. With the throttle lever in the FLX / MCT position the EEC will receive and convert the demanded thrust into an N1 Primary Thrust.

Fixed de-rated settings are selected within the ACTIVE submenu of the FMS; select the PERF page to select the DERATED, D1, D2, D3 etc. requirement.

The engines have the capability of applying fixed percentage thrust derate levels during take-off thrust on the Trent XWB-84. The following fixed derate levels apply:

- Take-Off Derate Level 1 (4%)
- Take-Off Derate Level 2 (8%)
- Take-Off Derate Level 3 (12%)
- Take-Off Derate Level 4 (16%)
- Take-Off Derate Level 5 (20%)
- Take-Off Derate Level 6 (24%)

Power Setting - Basic Control Loop

The engine thrust levers send Throttle Control Assembly (TCA) signals to the Propulsion Control System (PCS). The Throttle Potentiometer Angle (TPA) is then processed together with information such as altitude, Mach number and ambient conditions to calculate the required thrust setting value for the engine condition. This value can be either Primary N1 thrust mode, or if sufficient air data is not available N1 command in alternate mode.

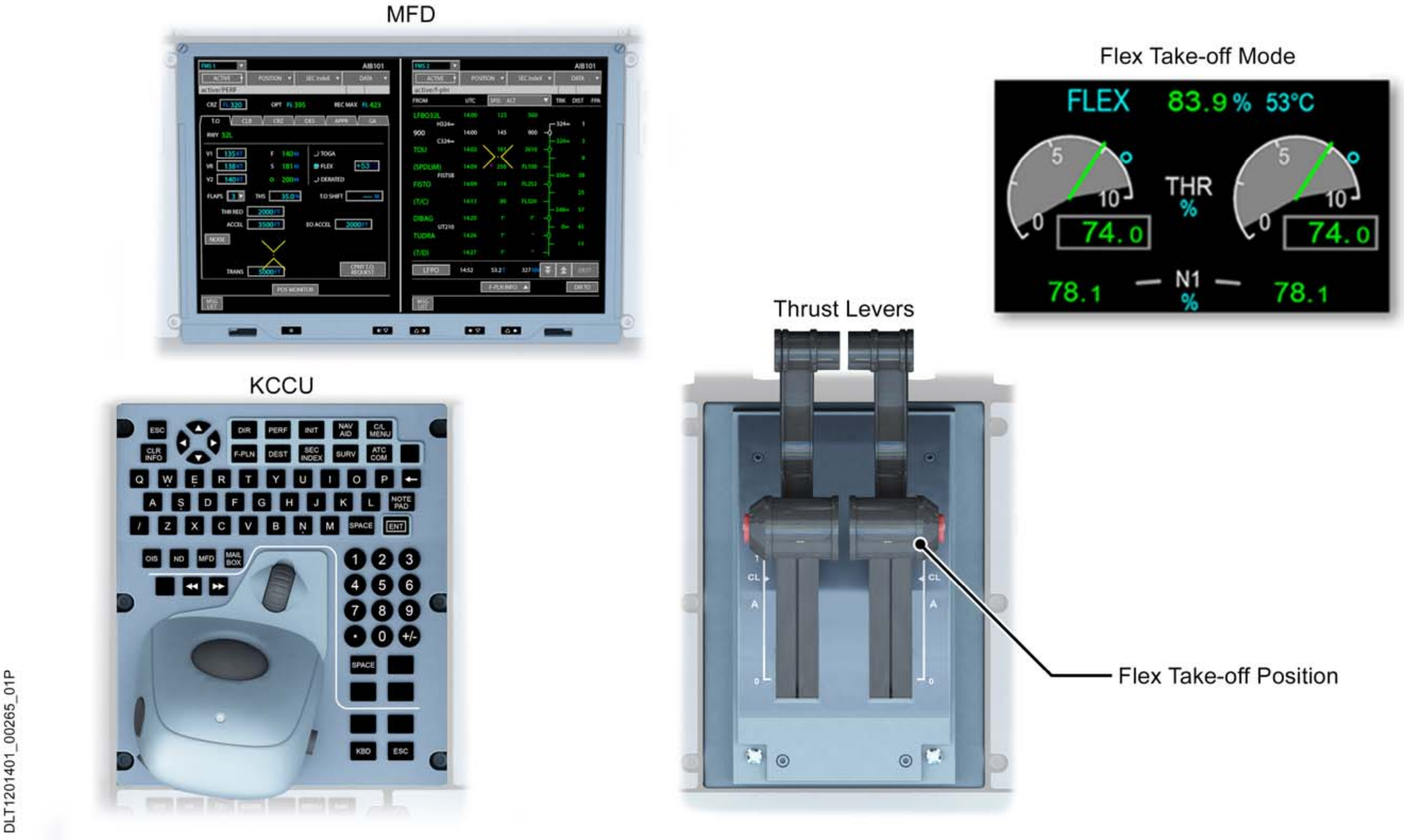
The commanded output of Primary N1 is transmitted to the engine accessory mounted control units e.g. EEC, HMU.

The main unit associated with fuel control is the Hydro-Mechanical Unit (HMU). Inside the HMU a fully modulating Fuel Metering Valve (FMV) is used to regulate the amount of fuel sent to the combustion chamber.

The position of the FMV is controlled using a dual wound FMV torque motor that is connected independently to each channel of the EEC, but driven only by the controlling EEC channel.

The position of the FMV is continuously indicated to both channels of the EEC by a dual wound Linear Variable Differential Transducer (LVDT).

The EEC uses the feedback from the LVDT position to make small adjustments to the command signals in order to achieve the desired condition, a method of control commonly described as a closed loop control system.



FLEX TAKE-OFF / FIXED DERATE SETTINGS

Fuel System

Location

The locations of the fuel system components are on the external gearbox, LP compressor case and engine core.

Purpose

The purpose of the fuel system is to:

- Supply fuel to the engine with the correct properties for combustion
- Control flow necessary for ease of starting, acceleration and stable running
- Limit exposure of the HMU to sub-zero fuel temperatures
- Provide motive (muscle) pressure for component actuation

Description

The fuel system components are:

- The Main Engine Fuel Pumps (MEFP)
- The Fuel Oil Heat Exchanger (FOHE)
- LP fuel filter
- Hydro-Mechanical Unit (HMU)
- Fuel Flow Transmitter (FFTX)
- HP fuel strainer
- Fuel Spray Nozzles (FSN)
- LP Fuel Filter Differential Pressure Transducer
- Fuel Temperature Sensor
- Fuel Drains Tank

The fuel system can be divided into sub-systems:

- Supply
- Control / protection.

Supply

To satisfy demand, the supply system must be able to:

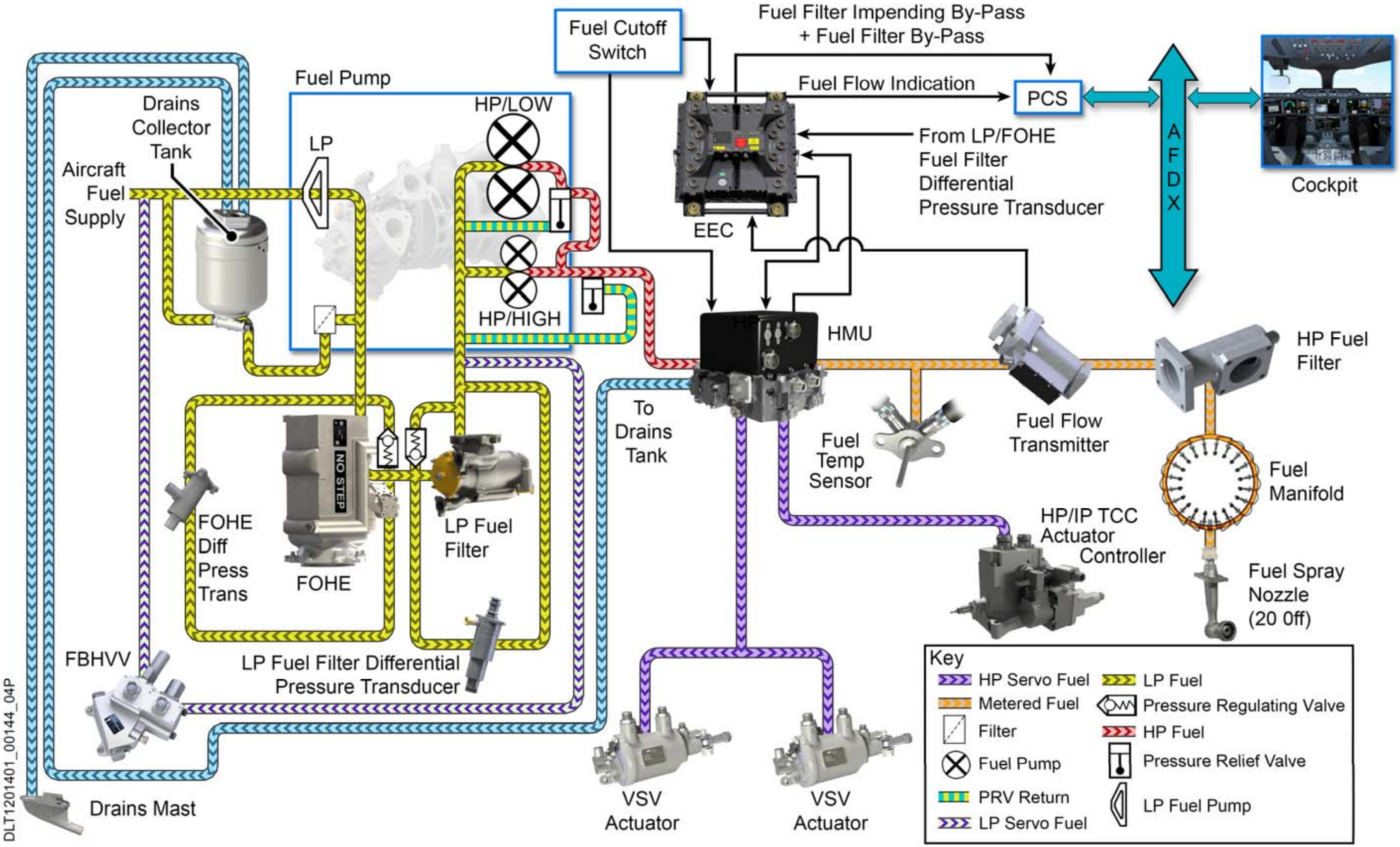
- Increase aircraft fuel delivery pressure to overcome the pressure drop through the low-pressure system
- Limit exposure of the HMU to sub-zero fuel temperatures
- Remove contaminants from the fuel
- Increase the pressure of the fuel delivered to the combustor to ensure satisfactory atomisation before ignition

Control / Protection

The aircraft and FADEC system interface with the HMU to control the fuel through all operating conditions.

The fuel control system;

- Meters the fuel delivered to the combustion chamber to satisfy all engine thrust requirements
- Manually shuts-off of the fuel flow to the combustor by the flight crew
- Automatically shuts-off of the fuel flow to the combustor by the FADEC system
- Removes any remaining fuel from the fuel manifold on engine shutdown (on the ground only). In flight the manifold lines remain full for quick relight if required.



FUEL SYSTEM

ENGINE FUEL SYSTEM

Main Engine Fuel Pump (MEFP)

Location

The MEFP is located on the right side of the external gearbox rear face.

Purpose

The purpose of the MEFP is to:

- Deliver sufficient fuel flow and pressure for efficient atomisation of fuel at the FSNs
- Provide sufficient fuel flow to operate any fuel-powered actuators
- Deliver demanded fuel flow in the event of failure of the aircraft fuel pumps

Description

The MEFP is a single LRU assembly consisting of low and high-pressure pumps and are driven by a common splined drive shaft which connects to the external gearbox drive train.

The pump is designed to be capable of providing sufficient fuel flow to satisfy the demanded burned and off take flows for maximum take-off fuel demand and also for windmill relight fuel demand at low HP spool speed.

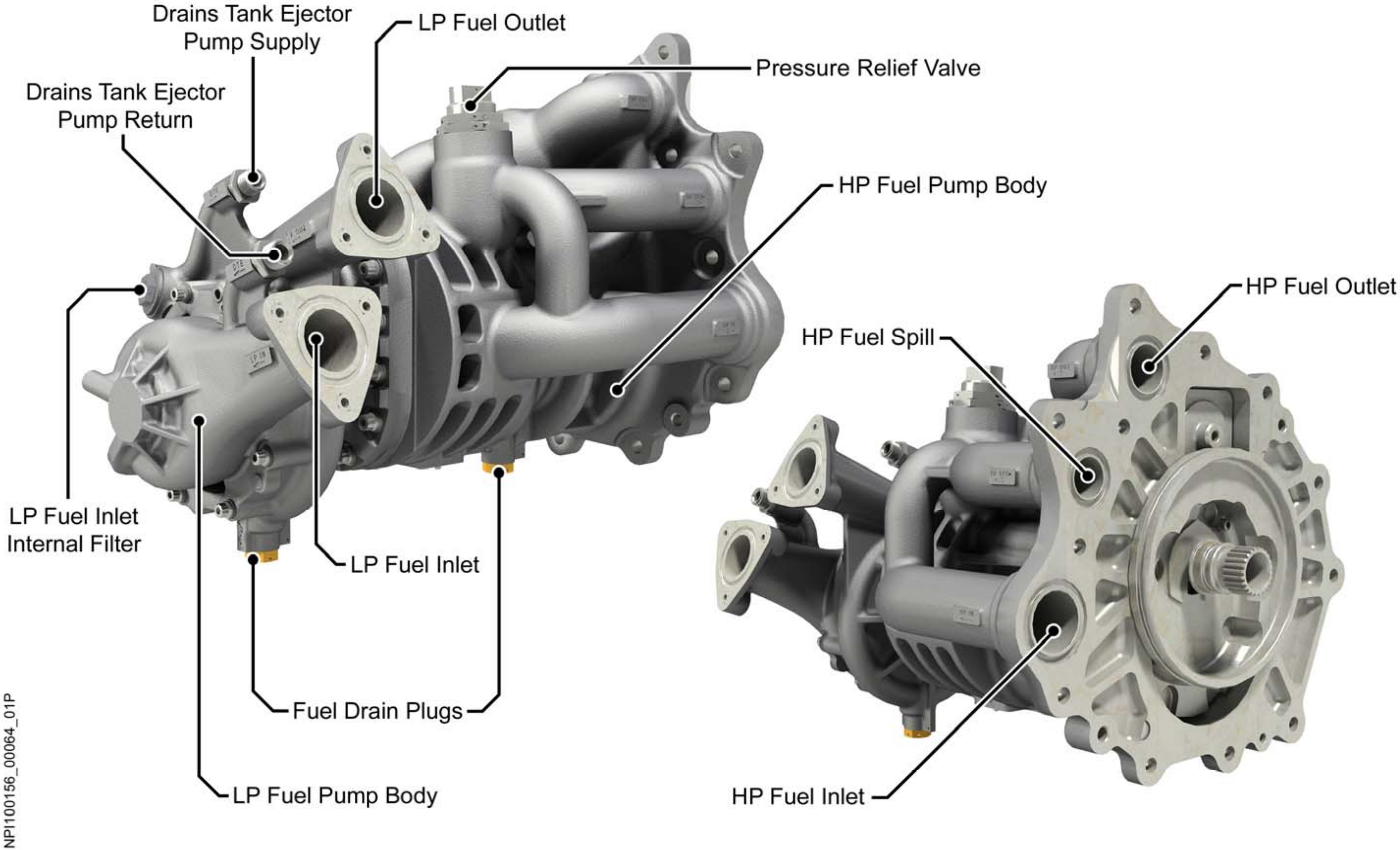
The centrifugal LP pump provides sufficient pumping capacity and pressure rise to allow for adequate filling of the HP gear stages after accounting for interstages pressure losses. The size of the LP pump is set so that the LP delivery pressure is high enough to overcome the maximum (worst-case) pressure drop though the LP fuel filter while still meeting the requirements of the HP fuel system.

The LP pump is designed to be capable of meeting its requirements when exposed to fuel contaminated with solid contaminants and ice. In the case of ice contamination, the MEFP is required to satisfy the requirements when exposed to a high concentration transient release of ice.

The HP pump configuration has two positive displacement parallel spur gear pumps of differing sizes that supply fuel to the HMU during engine operation.

The small HP gear stage is designed to deliver fuel flow to the HMU in order to satisfy the burner flow and HP stage off take flows for all performance points up to and including normal cruise flow.

The large HP gear stage is designed to augment the fuel flow delivered by the small HP gear stage when operating at performance points above normal cruise in order to satisfy all burner flow and off take flow demands. The large HP gear stage is designed to spill its outlet flow to the HP pump inlet via a low pressure rise spill loop in the HMU when the small HP gear stage is capable of satisfying the demanded fuel flow in order to minimise fuel heating.



MAIN ENGINE FUEL PUMP (MEFP)

Main Engine Fuel Pump (MEFP)

Operation

Fuel is delivered from the aircraft fuel pumps via the low pressure shut off valves in the wing tanks to the engine MEFP.

Fuel enters the inlet of the single stage centrifugal impeller LP pump where the fuel pressure is raised (75psid at typical ground idle conditions) for delivery to the FOHE. In addition the LP pump pressure delivery, supplies motive power to operate both the Front Bearing Housing Vent Valve (FBHVV) and the Drains Tank Ejector Pump (DTEP) whose flow induces suction through the ejector valve of the drains tank to remove any fuel in the tank and returns it to the inlet of the LP pump.

LP pump delivery passes to the FOHE where the fuel is pre-heated in the FOHE and passed through the LP fuel filter, it returns to the MEFP and enters the HP pump stages inlet.

As described earlier, the small HP gear stage is designed to deliver fuel flow to the HMU in order to satisfy the burner flow up to cruise and HP stage off take flows for all performance points above normal cruise flow.

The large HP stage outlet flow is controlled by the position of a combined Non Return Valve (NRV) & Pressure Relief Valve (PRV) which is mounted in the MEFP. The position of the NRV / PRV is determined by the HMU Spill valve position. When fuel demand is satisfied by the small HP gear stage alone, the NRV / PRV is closed and the HP large gear stage output is routed to the HMU via the spill valve and returned to HP inlet.

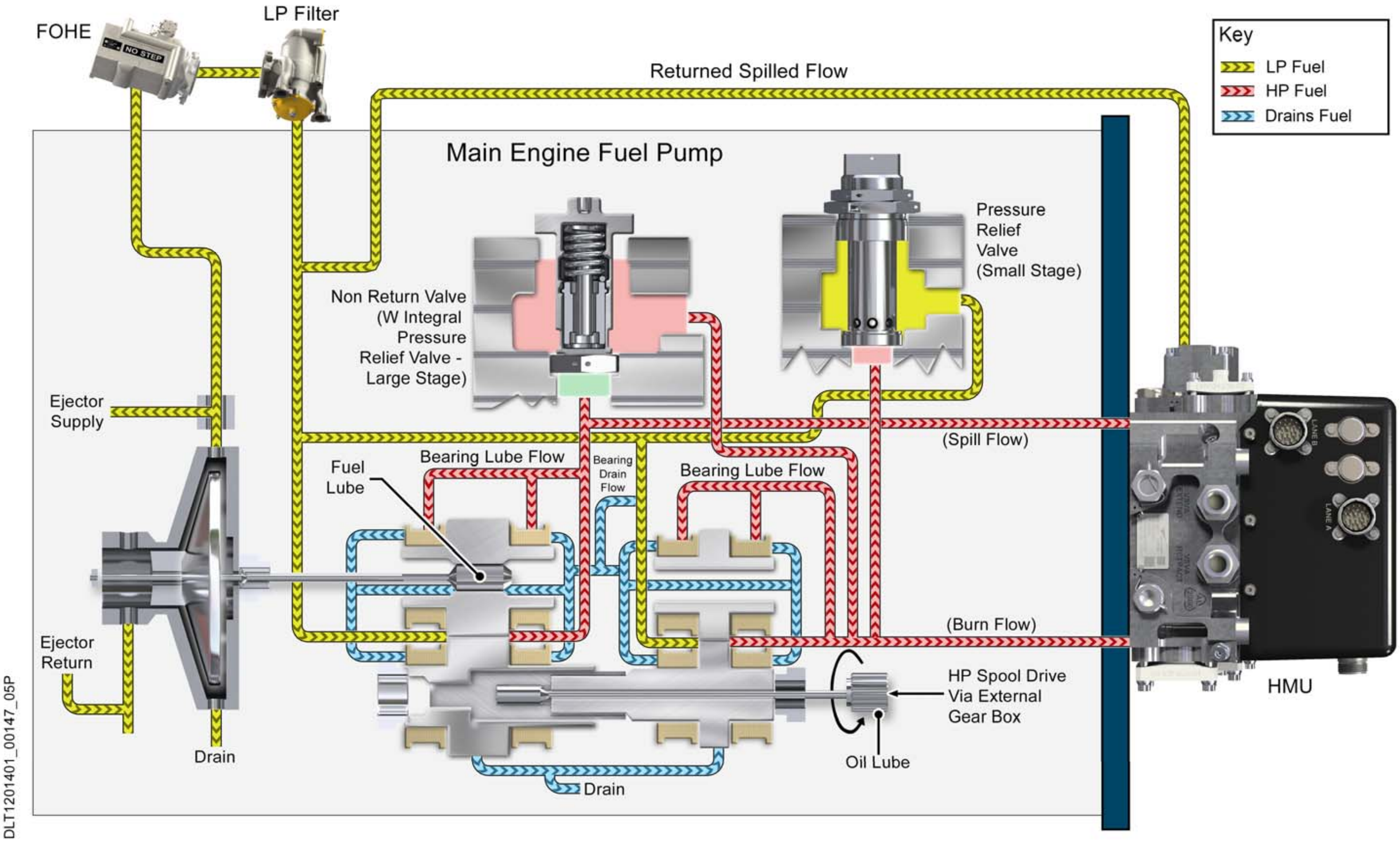
When fuel demand increases and cannot be met by the small gear stage alone, the HMU spill valve return flow is reduced which has the effect of building pressure in the NRV / PRV to overcoming the spring pressure and HP pump down-stream pressure. Fuel delivery to the HMU is now increased by the large HP gear stage via the NRV / PRV.

The system is designed to deliver the demanded fuel flow during negative ‘G’ events and be capable of re-priming the fuel system when air is introduced.

The interface between the MEFP and HMU is through tubes and an adaptor block.

Also NRV / PRV limits the pressure rise of the pump in the event of system blockage.

On the MEFP body there are two drain plugs provided for draining fuel from the LP and HP fuel systems in order to facilitate removal of the MEFP or other fuel system unit.



MAIN ENGINE FUEL PUMP (MEFP)

Fuel Oil Heat Exchanger (FOHE) and LP Filter Housing

Location

The FOHE and LP Fuel Filter assembly are located on the right side of the LP compressor case.

Purpose

The purpose of the FOHE is to exchange heat within the oil and fuel circuits to maintain the oil and fuel temperatures within acceptable limits. The LP fuel filter removes any contaminants from the LP fuel prior to delivery to the HP pumps and HP fuel components.

Description

The FOHE has been described in the oil system. These notes describe the FOHE from a fuel perspective.

The FOHE allows heat to be transferred from the oil to the fuel to reduce the possibility of the water content in the fuel from forming ice crystals.

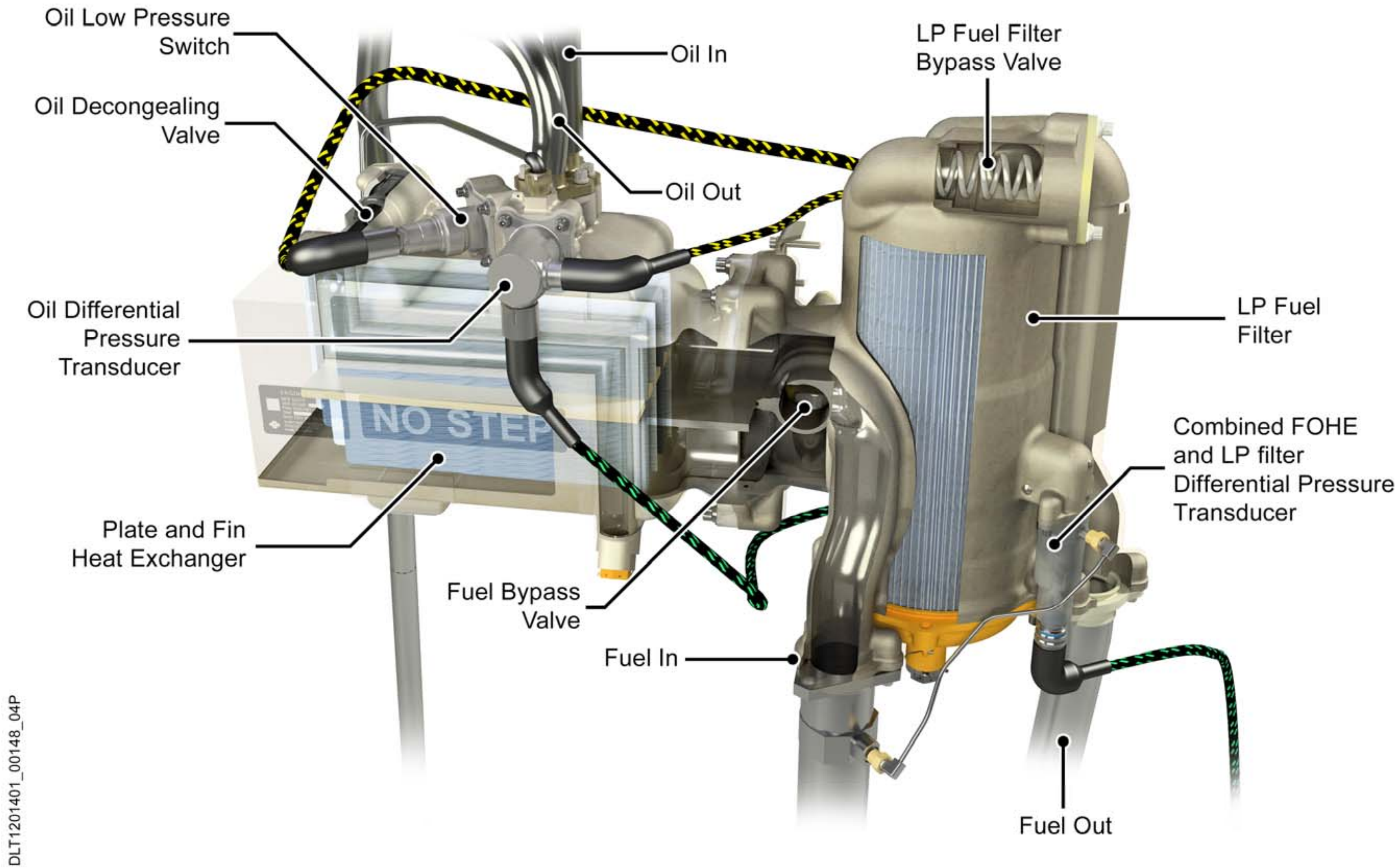
The FOHE matrix is designed using a number of plates separated by fins through which the fuel flows in an opposite direction to the oil to ensure maximum heat transfer between the oil and fuel.

Operation

The fuel enters the FOHE matrix from the LP pump and flows through internal passages before delivery to the LP filter housing.

At the inlet of the FOHE, and included within the FOHE design, is a passive fuel bypass valve (35 psid). The valve prevents and limits unwanted engine fuel flow and fuel starvation in the event of sudden release of ice (snow-shower) from the aircraft/engine fuel system to the FOHE inlet.

The LP fuel outlet from the FOHE passes to the LP fuel filter housing, where a 40 micron, high temperature, ribbed, corrugated, impregnated element filters the fuel prior to delivery to the HP stages. The flow path within the filter housing is from outside to inside. Should the LP fuel filter become blocked, a bypass valve (25 psid) located within the filter housing, will open to deliver unfiltered fuel to the HP pump stages.



LP FILTER & FUEL OIL HEAT EXCHANGER

LP Fuel Filter / FOHE Differential Pressure Transducer

Location

The combined LP fuel filter / FOHE differential pressure transducers are attached to LP fuel filter element housing.

Purpose

For filter / FOHE blockages the transducers detect four levels of blockage and provides the following associated signals to the airframe;

- LP Filter- impending
- LP Filter- pre-bypass
- LP Filter- bypass
- FOHE - bypass

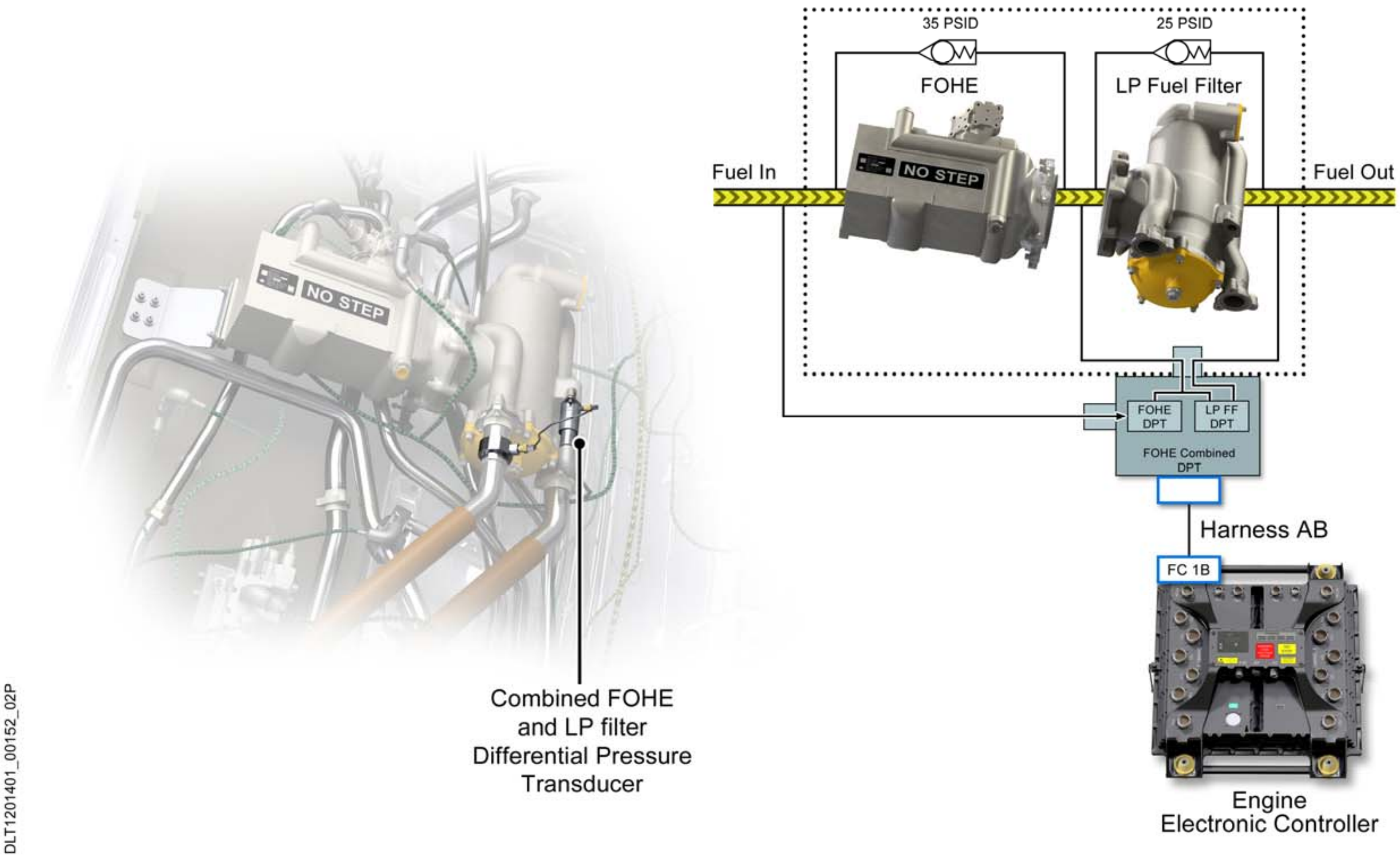
Description

A differential pressure transducer is mounted on the engine fuel oil heat exchanger and reads the pressure across the Low Pressure fuel filter element of the FOHE to enable detection of the build-up of debris across the filter.

Whilst the FOHE is designed to cope with the most arduous non blocked operating conditions in terms of iced fuel, it also features a fuel side bypass valve to ensure that no fuel flow restriction can occur should high concentrations of ice be released from the aircraft fuel system, which could block the front face of the FOHE matrix. The valve is designed to operate when the fuel side of the unit becomes blocked such that the pressure drop exceeds a given threshold.

Differential pressure signals are sent to the EEC by a twin sensor transducer (double element sensor with simplex output per element sensor).

The warnings are displayed in the cockpit.



LP FUEL FILTER DIFFERENTIAL PRESSURE TRANSDUCER

Hydro-Mechanical Unit (HMU)

Location

The HMU is located to an adaptor block on the right side of the external gearbox.

Purpose

The purpose of the HMU is to follow the commands of the EEC to meter the fuel to the FSNs, under all flight envelope conditions. The HMU also provides motive fuel servo pressure for the VSV / TCC actuators, and fuel shut off either manually, automatically or during emergency conditions.

Description

Various electro-hydraulic servo valves provide HP fuel servo pressure to operate internal valves or external components.

Variable Stator Vane (VSV) Servo valve

The VSV servo valve is connected to both channels of the EEC through electrical harnesses which provides variable electrical current to change the position of the servo valve. The subsequent hydro-mechanical movement affects the HP fuel servo pressure and therefore the VSV actuators position.

Metering Valve Servo valve

The HMU receives commands from either channel of the EEC through electrical harnesses to command a servo valve inside the HMU. The servo valve directs HP servo pressure to control the position of the metering valve. The metering valve in conjunction with the Pressure Drop Control Valve (PDCV) and the combined spill valve controls the fuel flow to the fuel spray nozzles. A Linear Variable Differential Transducer (LVDT) feedback device provides the EEC with signals of valve position.

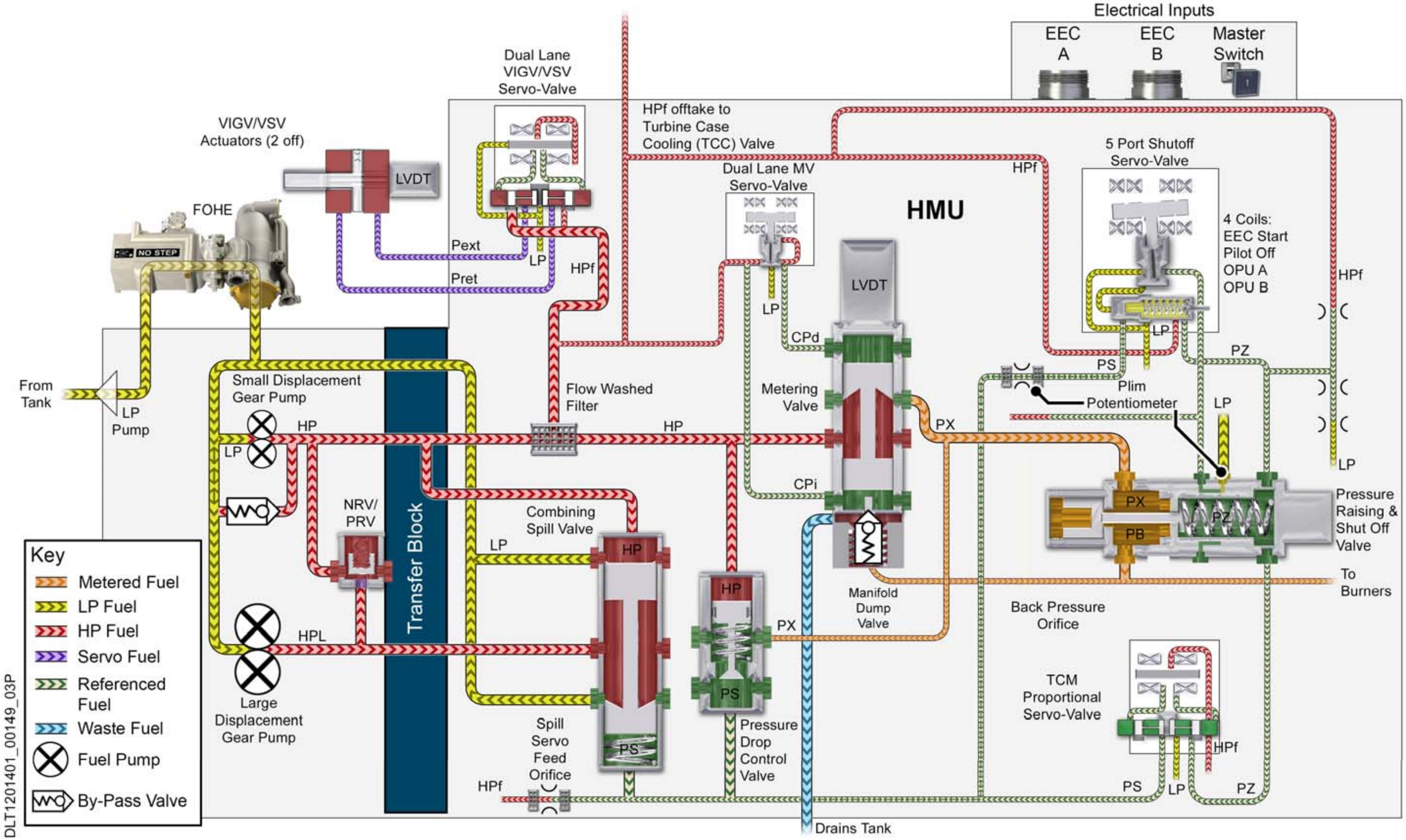
A fuel manifold dump valve is mechanically connected to the metering valve to drain the fuel manifold into the drains tank when the engine is shut down. This operation is only enabled when the aircraft is on the ground.

Shut-off Servo valve

The shut-off servo valve directs HP servo pressure to control the position of the Pressure Raising and Shut off Valve (PRSOV). The shut-off servo valve is connected directly to the Engine Master Start switch (Eng 1 or 2) on the cockpit ENG START panel, to either command fuel to the fuel spray nozzles on start up or to shut the fuel off during engine shut down. The shut-off servo valve is also connected to the engine protection systems in the EEC to shut the engine down in case of LP turbine overspeed, LP rotor overspeed and IP rotor overspeed.

Thrust Control Malfunction (TCM) Servo valve

The TCM function in the EEC interfaces with the TCM servo valve in the HMU to; reduce engine thrust, shut the engine down or cap the engine speed in accordance with the limits of the flight envelope. TCM servo valve using servo pressure controls of the position of the PRSOV, the PDCV and the combined spill valve. This valve operates independently to the metering valve control, but can modulate flow as demanded by the EEC in the event that control of the metering valve is lost.



HYDROMECHANICAL UNIT (HMU)

Drains Tank and Ejector Valve

Location

The fuel drains collector tank is located on the right side of the LP compressor case.

Purpose

The purpose of the drains tank is to collect the fuel remaining in the fuel manifold after a usual engine shutdown on the ground or after a failure to start in order to minimise the build-up of carbon in the fuel spray nozzles and lacquering of the fuel manifold tubes.

Description

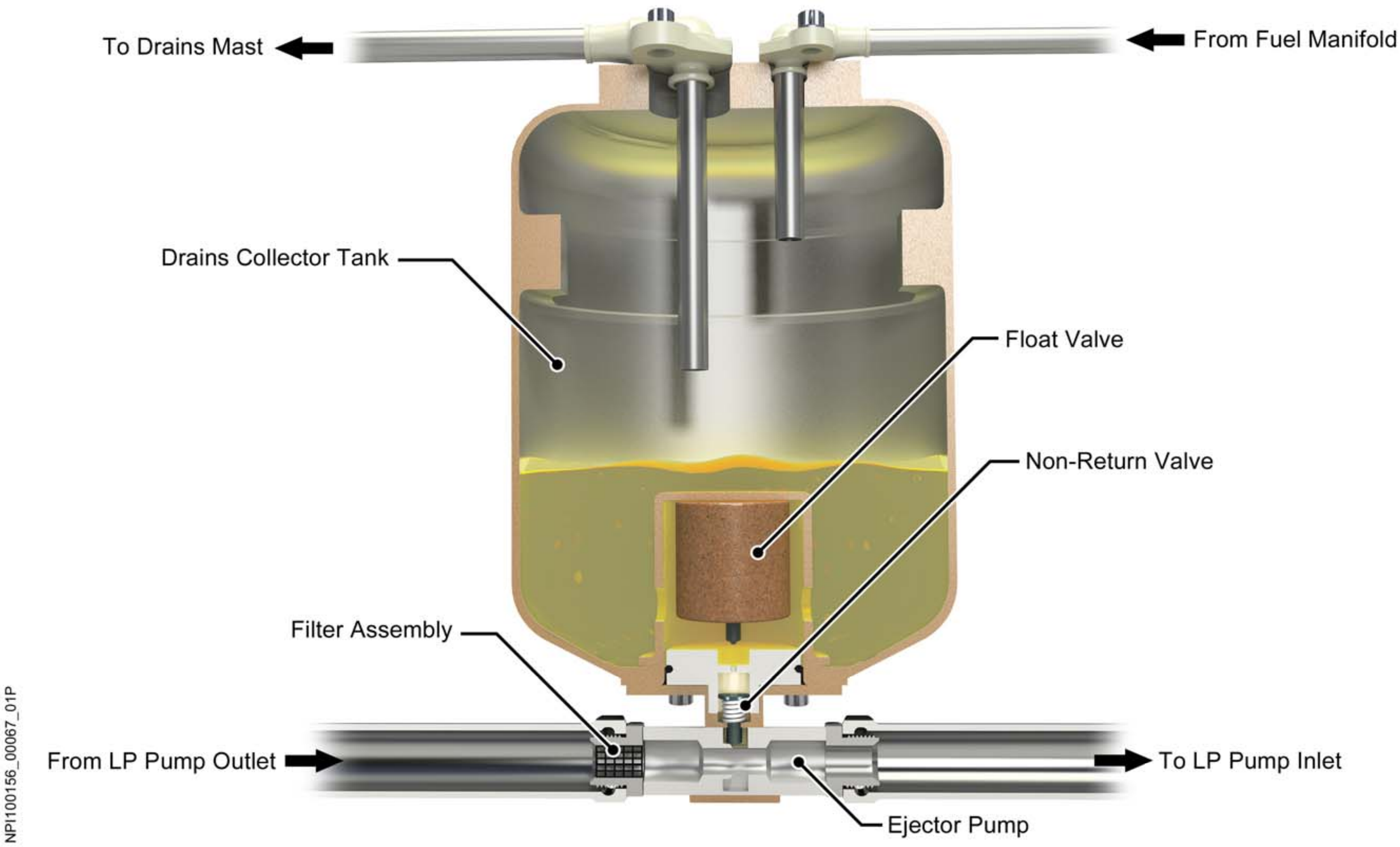
The drains tank assembly consists of a steel tank and an ejector valve. Within the ejector valve is a non-return valve, a float valve and a filter assembly.

The Fuel Spray Nozzles (FSNs) are connected to, and supplied with fuel from, a fuel manifold. During engine shutdown on the ground only, the drains collector tank receives fuel from the tubes between the HMU and fuel manifolds through a valve located within the hydro-mechanical unit (HMU).

During the next engine start fuel from the low pressure fuel pump passes through the ejector pump at the bottom of the drains collector tank. By a venturi effect any fuel in the drains collector tank is returned to the inlet of the LP fuel pump. As the level of fuel in the tank reduces the float valve will close off the inlet to the non-return valve allowing the non-return valve to close and preventing air from entering the fuel system.

If an in-flight shutdown happens the valve in the HMU remains closed keeping the fuel in the tubes and manifolds up to the FSNs in readiness for an in-flight start.

The drains tank collector is sized to hold fuel for one usual engine shutdown and a number of failed starts. If the number of failed starts is more than the maximum limit the drains tank can become full and the fuel will be dumped overboard through the drains mast.



DRAINS TANK AND EJECTOR VALVE

Fuel Flow Transmitter

Location

The fuel flow transmitter is located in the main HP fuel tube between the HMU and the HP fuel strainer.

Purpose

The purpose of the flow transmitter is to provide a measurement of the mass rate of fuel to the fuel spray nozzle manifold.

Description

The fuel flow transmitter interfaces with the EEC through an electrical harness. The transmitter provides a signal to the EEC that is the same frequency as the mass fuel flow to the fuel spray nozzle.

The EEC converts the frequency of the signal into units that are recognised by the flight crew and the aircraft systems and transmits the information through the AFDX system to the ECAM as a secondary parameter.

The flow transmitter output can be configured to specific Customer requirements to read in Fuel Flow units of ‘lbs’ or ‘kg’.

Fuel Temperature Sensor

Location

The fuel temperature sensor is located in the fuel tube between the HMU and the fuel flow transmitter.

Purpose

The purpose of the fuel temperature sensor is to sense the temperature of the fuel for use by the:

- Engine Heat Management System
- Engine Health Monitoring

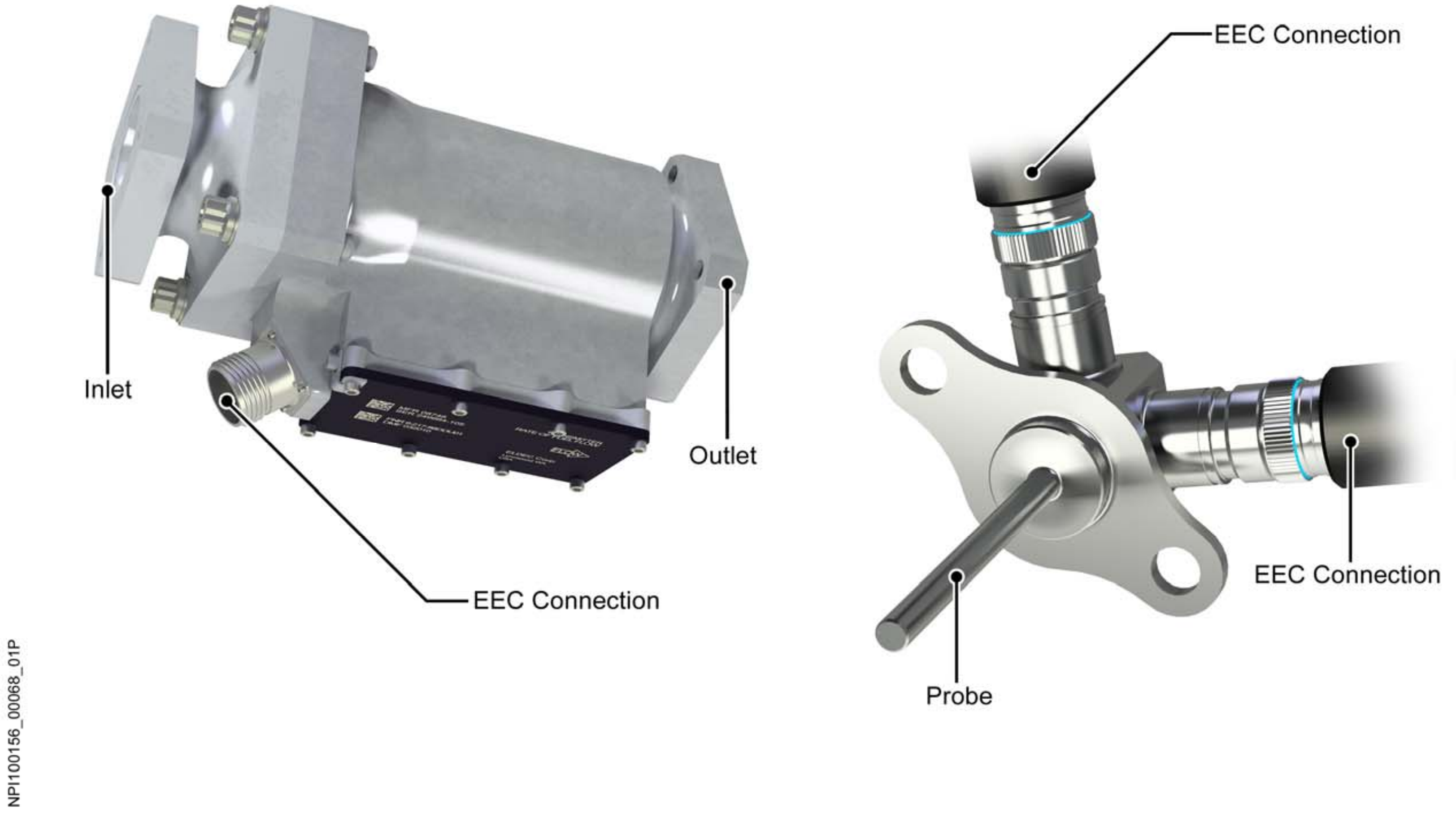
Description

The fuel temperature sensor is a dual channel resistive temperature device (RTD) that interfaces with both channels of the EEC through electrical harnesses.

The EEC uses fuel temperature to manage the temperature of the fuel and oil as part of the heat management system, keeping the lubricating oil cool and the fuel to be kept warm.

If fuel temperature increases above a maximum threshold (currently defined as 100°C), the Oil Bypass Valve (OBV) closes and the oil is forced through the SAOHEs. This lowers the oil temperature and as a consequence the fuel temperature.

The minimum HP fuel temperature at which the OBV should be closed is 40°C which in turn corresponds to a 20°C LP fuel filter inlet fuel temperature. This ensures that the LP fuel filter inlet temperature remains above 0°C.



FUEL FLOW TRANSMITTER / TEMPERATURE SENSOR

HP Fuel Strainer (HPFS)

Location

The HP fuel strainer is located at the inlet of the fuel manifold at the six o'clock position.

Purpose

The purpose of the HP fuel strainer is to protect the fuel spray nozzles against contamination.

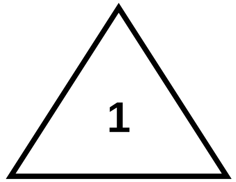
Description

The HP fuel strainer is secured by a single captive bolt within the strainer and bolted to the filter housing assembly of the fuel manifold.

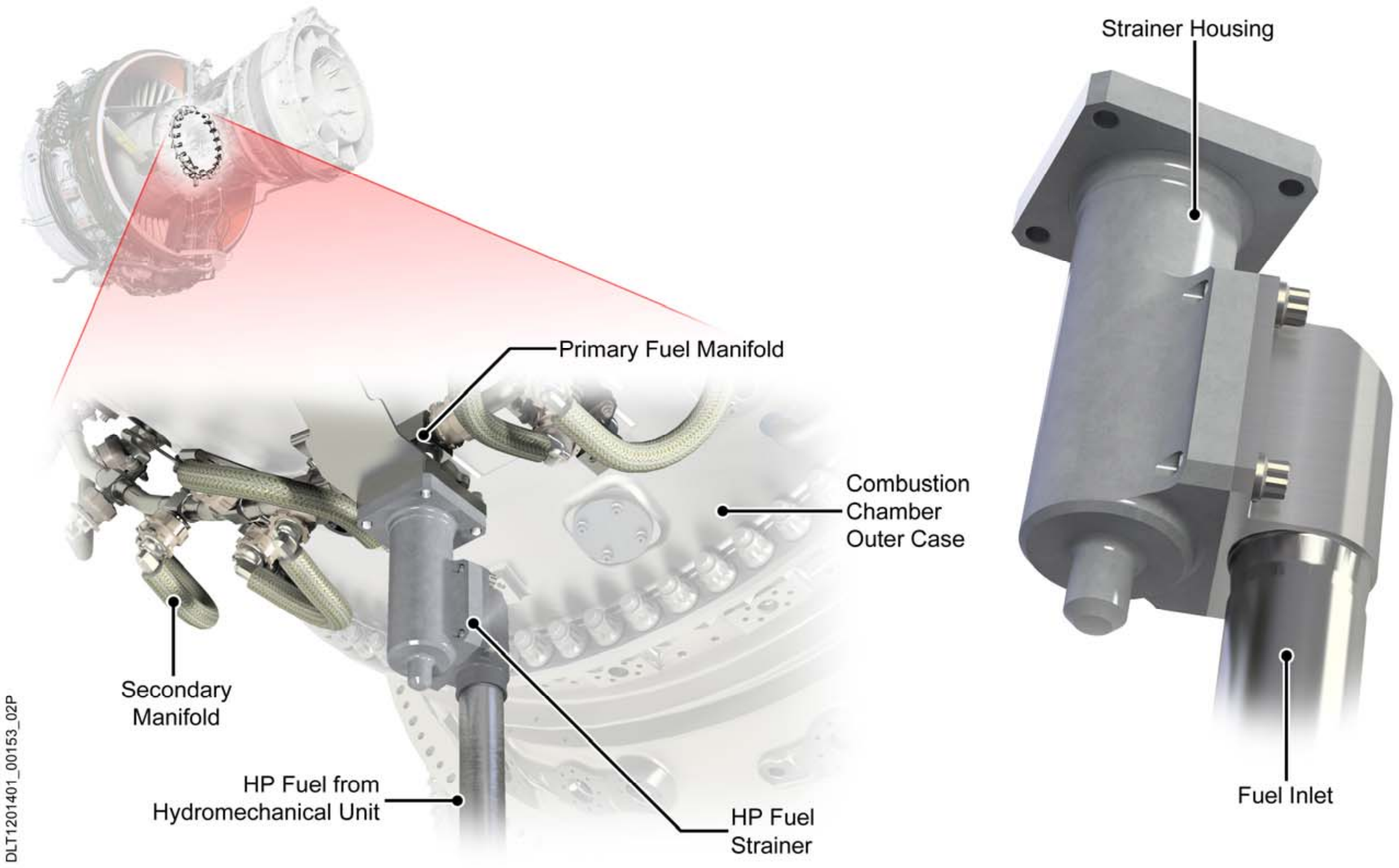
The HP fuel strainer provides protection against blockage of the fuel spray nozzles. The strainer is constructed from a plain weave stainless steel mesh with a 250-micron mean pore opening. The strainer is cleanable by washing in kerosene or other AMP approved fluid. The strainer must be discarded after two cleaning process. Once cleaned the strainer must be marked with the appropriate markings as stated in the AMP cleaning procedure (TRENTXWB-A-73-11-12-01A01-250A-A).

Check the number of times the strainer has been cleaned. The number of triangles with a number 1 inside indicates this as follows:

- If there are two (2) triangular marks then discard the strainer.
- If there are less than two (2) triangular marks, clean the strainer.
- If no contamination is found engrave a triangle with a number One (1) inside (as below) to indicate the strainer has been cleaned.



The HP fuel strainer attachment bolt, must be secured with the correct torque loading (110 lbf.in) and always in accordance with the current the AMP task instructions.



HP FUEL STRAINER ELEMENT

HP FUEL STRAINER ELEMENT

Fuel Manifolds

Primary and Secondary Fuel Manifolds

Location

The primary fuel manifold is attached around the outside of the outer combustion case by brackets. Secondary fuel manifolds connect each fuel spray nozzle to the primary fuel manifold.

Purpose

The purpose of the primary fuel manifold is to deliver metered and filtered high-pressure fuel to the secondary manifold for the fuel spray nozzles.

Description

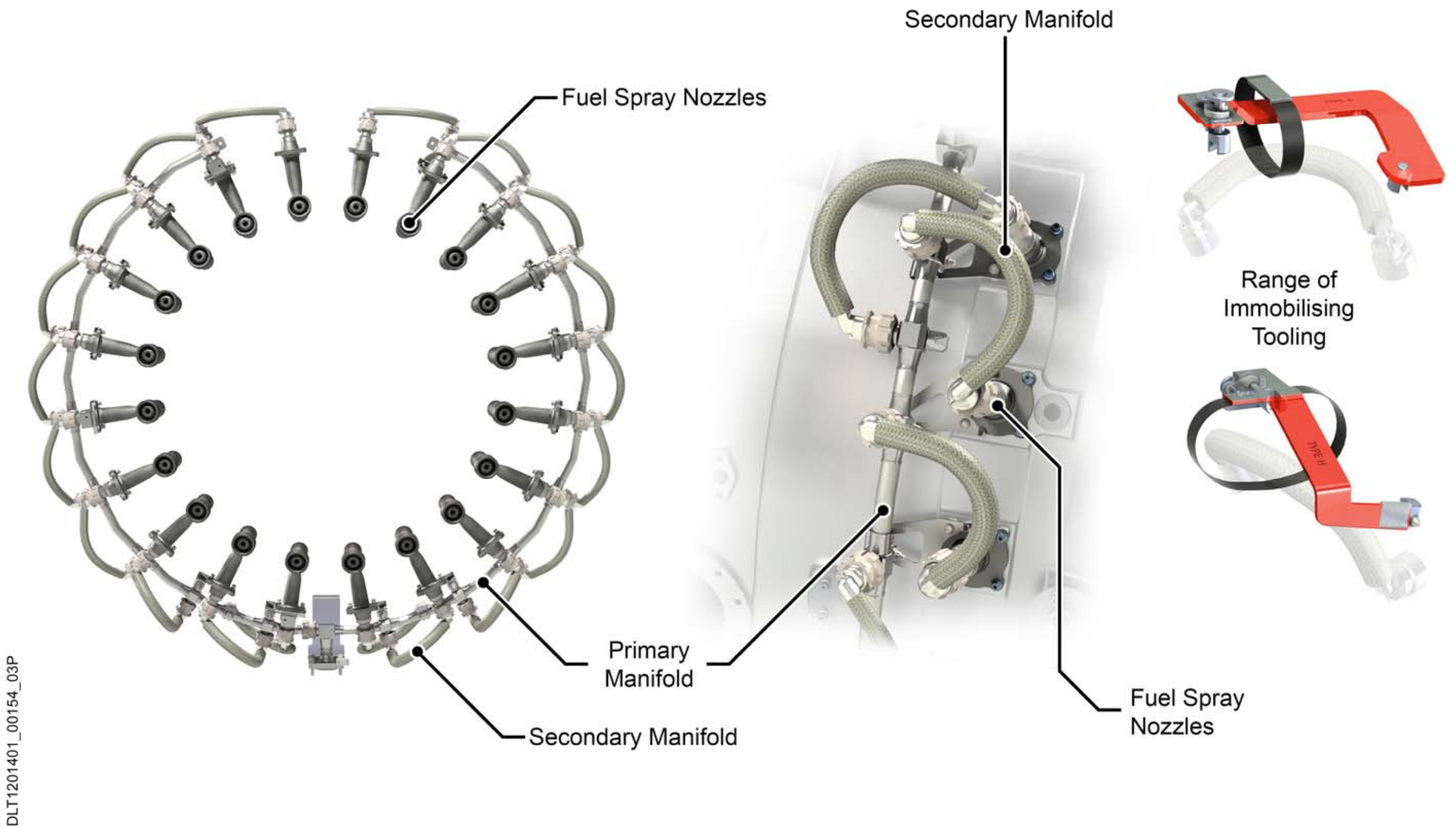
The primary manifold is a ridged tube that is manufactured and assembled in two halves around the outer combustor case.

Twenty flexible secondary manifolds allow fuel to flow to the fuel spray nozzles. Each secondary manifold delivers fuel to a single fuel spray nozzle.

The fuel manifolds are drained of fuel when the engine is shut down on the ground only, avoiding the potential for fuel inside the manifolds becoming heated during engine shutdown. The effect of which would be to cause lacquering in the fuel tubes and coking in the fuel spray nozzles, both of which have an impact on engine performance.

The secondary manifold hose configurations must be aligned and secured with the designated immobiliser tooling, during removal and installation.

The immobilisers ensure the correct orientation of the hoses and union angles to prevent any chaffing or potential HP fuel leakage due to incorrect installation.



FUEL MANIFOLDS

Fuel Spray Nozzles (FSN)

Location

The fuel spray nozzles are located through openings in the outer combustion case.

Purpose

The purpose of the fuel spray nozzles is to deliver metered air and atomised fuel to the combustion chamber.

Description

The twenty fuel spray nozzles are an internal heat shielded design with a coiled fuel tube. They each receive fuel through an individual flexible secondary fuel manifold. Fuel passes through body (feed arm) of the FSN to the swirl chamber where vanes swirl the fuel before it is atomises into a cone shaped spray.

Outer swirl vanes attached to the nozzle head allow HP compressor stage 6 exit air (P30) to mix with the fuel and help shape the flame in the combustion chamber.

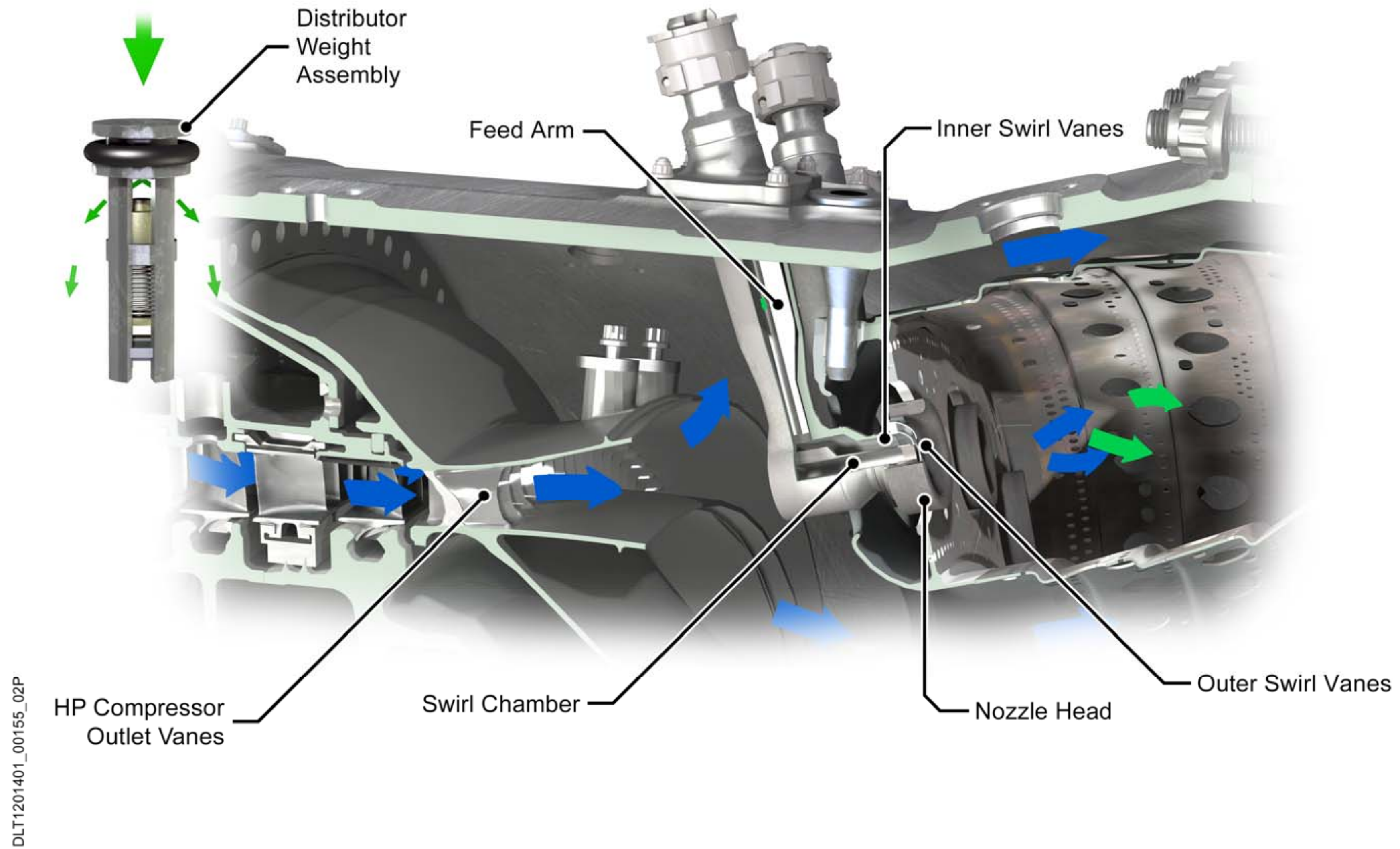
Other holes in the combustor head and dilution holes in the liners provide air into the combustion chamber to help shape the flame, insulate and cool the combustion chamber.

A distributor weight assembly is located inside the inlet of each of the FSN to control the fuel delivery pressure to the nozzle head at low fuel flow conditions.

The assembly consists of a housing, spring and weight. There are two different weights configurations depending on position, the nozzles which align with the igniters (8 & 12) are made from steel and identified by a straight line on the top of the distributor weight assembly (-), while all the others are made from tungsten and are identified by a plus on the top of the distributor weight assembly (+).

The weight configuration provides more fuel to the igniter nozzles at light up and helps ignition. The spring and weight are designed to open at a specific fuel manifold pressure.

This allows fuel to enter each FSN at the same time and to be lit evenly during starting and re-light and will shut off fuel during Master Start switch to ‘Off’ preventing carbon build-up on the FSN head.



FUEL SPRAY NOZZLE (FSN)

Maintenance Practices

LP Fuel Filter Element – Fuel Filter (73-11-11, 01-250)

Removal Procedure

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

LP Fuel Filter Element Removal

AMP Task

TRENTXWB-A-73-11-11-01A01-520A-A,

TRENTXWB-A-73-11-11-01A01-520A-D

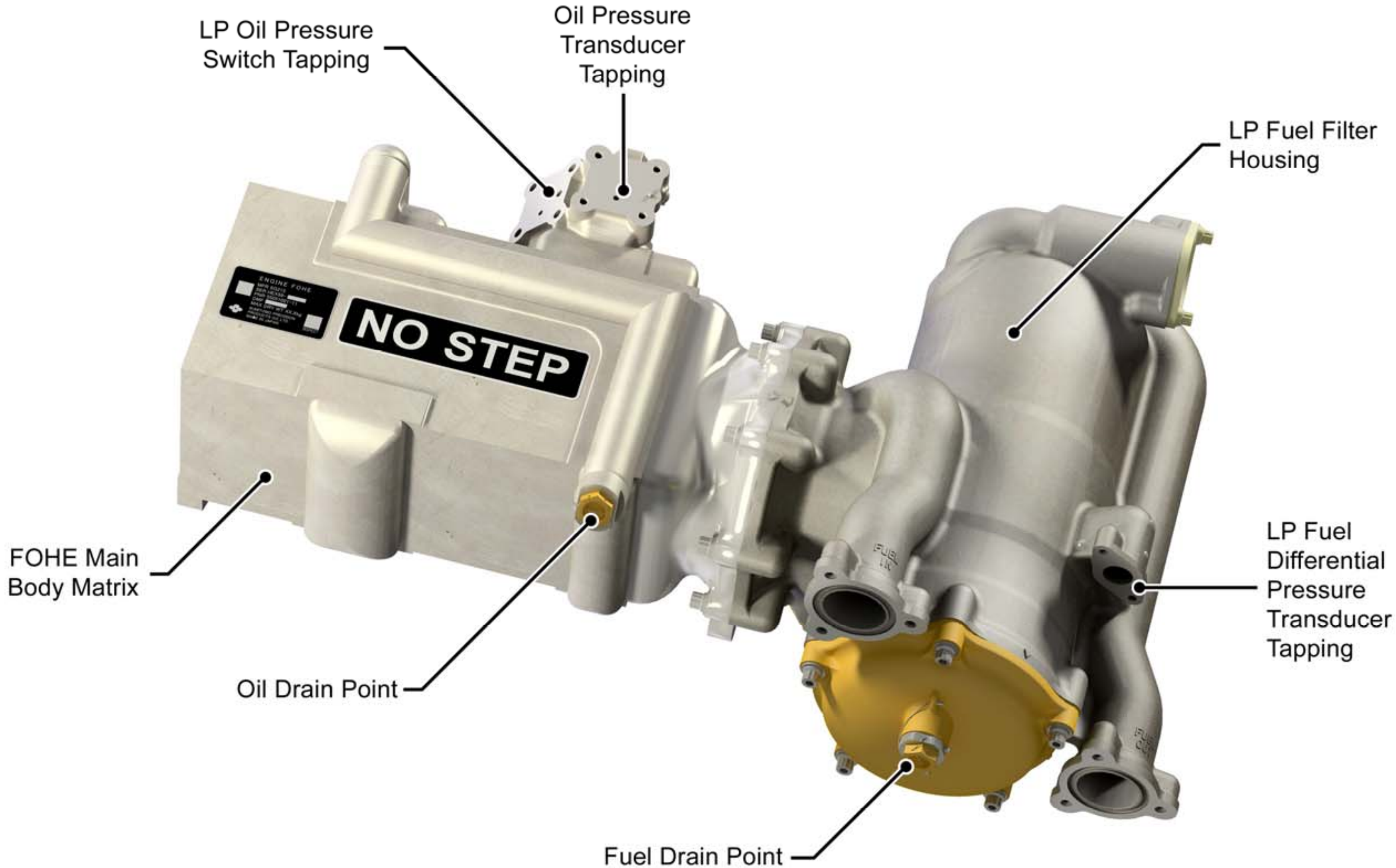
WARNING
BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING
YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE AN INJURY. IF YOU GET AN INJURY, PUT IT IN COLD WATER FOR 10 MINUTES AND GET MEDICAL AID.

WARNING
YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION
YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.

CAUTION
YOU MUST NOT LET ENGINE FUEL FALL ON THE ENGINE. REMOVE UNWANTED FUEL IMMEDIATELY WITH A CLEAN LINT-FREE CLOTH. THE FUEL CAN CAUSE DAMAGE TO THE SURFACE PROTECTION AND TO SOME PARTS.



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LP FUEL FILTER REMOVAL

Maintenance Practices

LP Fuel Filter Element – Fuel Filter (73-11-11, 01-250)

Installation Procedure

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

LP Fuel Filter Element Install

AMP Task

TRENTXWB-A-73-11-11-01A01-720A-A,

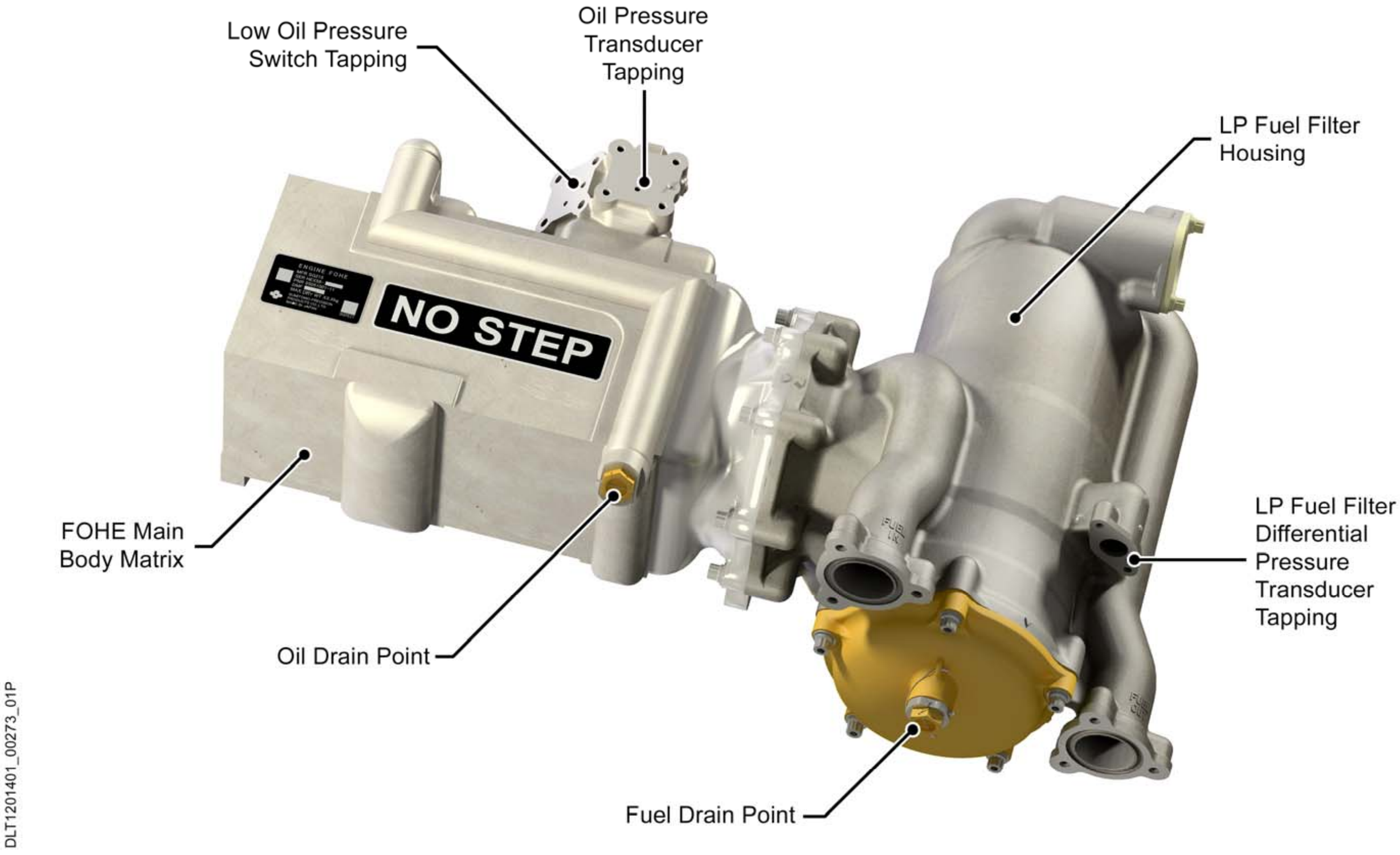
TRENTXWB-A-73-11-11-01A01-720A-D

WARNING

YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION

YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.



LP FUEL FILTER INSTALLATION

Maintenance Practices

HP Fuel Filter – HP Fuel Filter Element (73-11-12, 01-250)

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Remove Procedure.

AMP Tasks:

TRENTXWB-A-73-11-12-01A01-520A-A,

TRENTXWB-A-73-11-12-01A01-520A-D

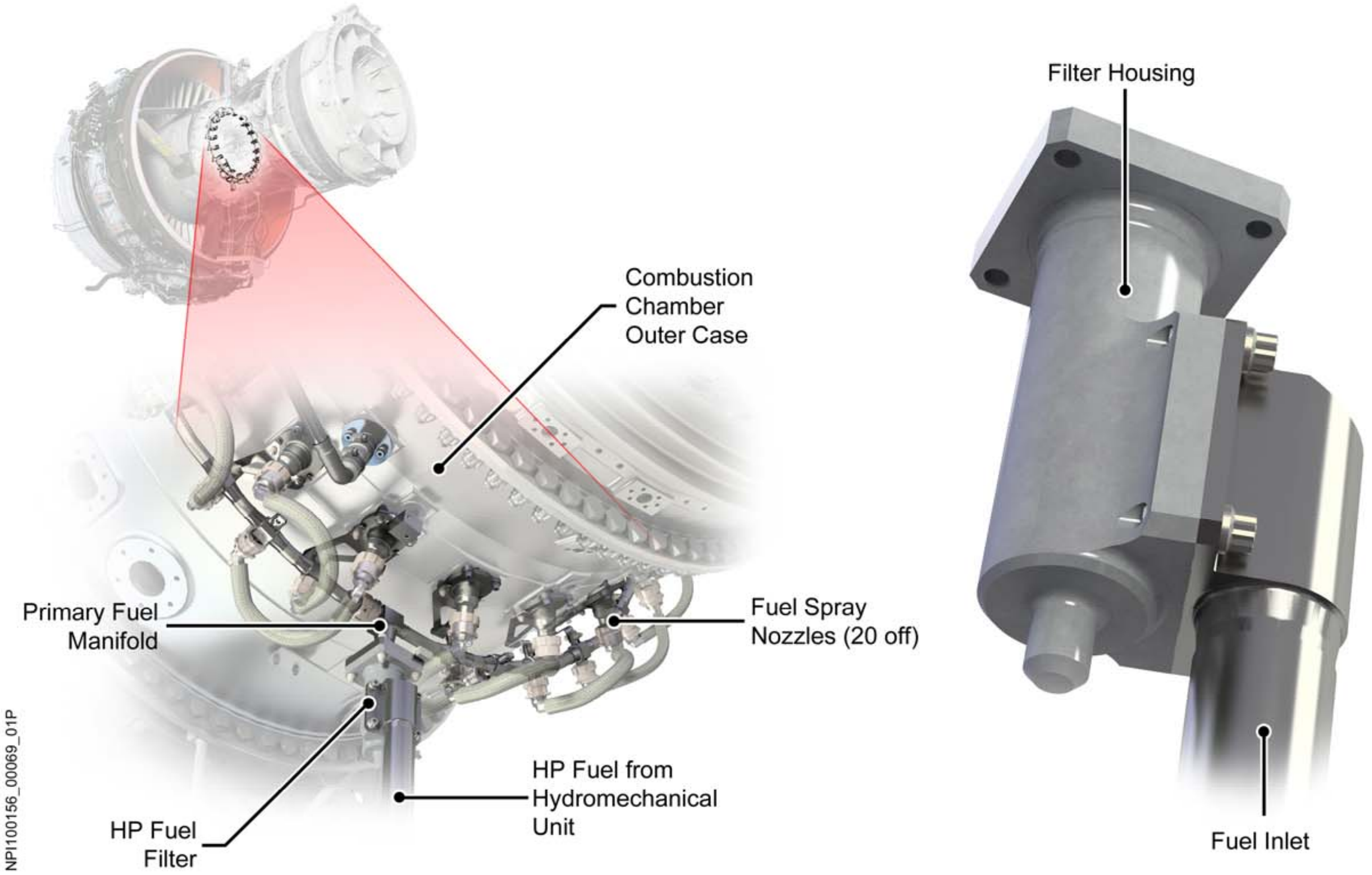
WARNING
BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING
YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE AN INJURY. IF YOU GET AN INJURY, PUT IT IN COLD WATER FOR 10 MINUTES AND GET MEDICAL AID.

WARNING
YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION
YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.

CAUTION
YOU MUST NOT LET ENGINE FUEL FALL ON THE ENGINE. REMOVE UNWANTED FUEL IMMEDIATELY WITH A CLEAN LINT-FREE CLOTH. THE FUEL CAN CAUSE DAMAGE TO THE SURFACE PROTECTION AND TO SOME PARTS.



HP FUEL FILTER REMOVAL

Maintenance Practices.

HP Fuel Filter – HP Fuel Filter Element (73-11-12, 01-250)

Clean and apply surface protection

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

AMP Tasks:

TRENTXWB-A-73-11-12-01A01-250A-A

(Ultrasonic cleaning)

TRENTXWB-A-73-11-12-01A01-250B-A

(Chemical cleaning)

WARNING

BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING

YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE AN INJURY. IF YOU GET AN INJURY, PUT IT IN COLD WATER FOR 10 MINUTES AND GET MEDICAL AID.

WARNING

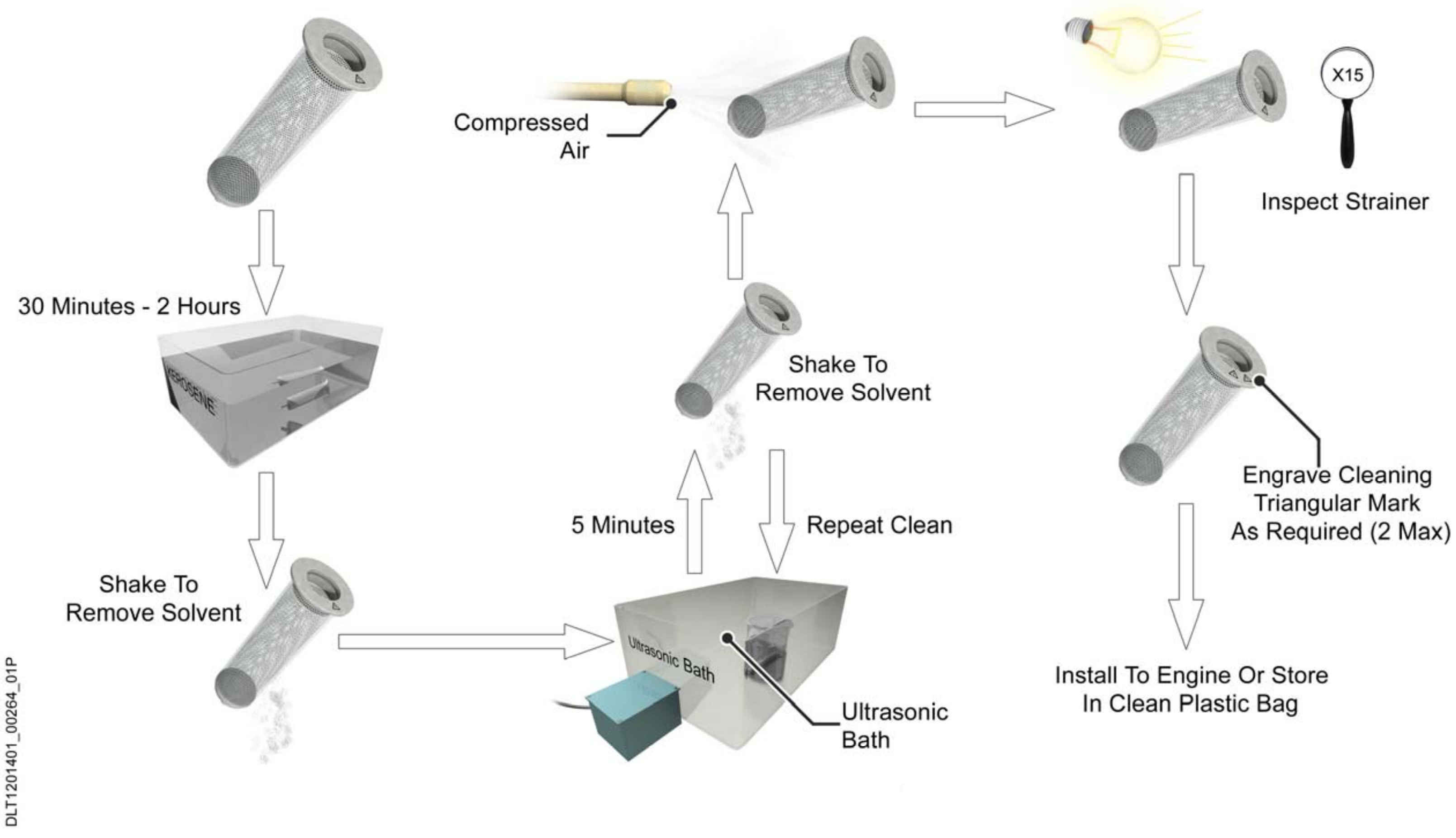
YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

WARNING

YOU MUST NOT GET KEROSINE ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. KEROSINE IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. KEROSINE FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET KEROSINE ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET KEROSINE IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION

YOU MUST NOT USE A COMPRESSED AIR SUPPLY OF MORE THAN 15 PSI (1.03 BAR) PRESSURE FOR A SPRAY GUN, OR TO DRY THE CLEANED PARTS. A COMPRESSED AIR SUPPLY OF MORE THAN 15 PSI (1.03 BAR) WILL CAUSE DAMAGE TO THE PARTS.



HP FUEL FILTER CLEANING

Maintenance Practices.

HP Fuel Filter – HP Fuel Filter Element (73-11-12, 01-250)

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

AMP Tasks: TRENTXWB-A-73-11-12-01A01-720A-A,

TRENTXWB-A-73-11-12-01A01-720A-D.

WARNING

BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING

YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE AN INJURY. IF YOU GET AN INJURY, PUT IT IN COLD WATER FOR 10 MINUTES AND GET MEDICAL AID.

WARNING

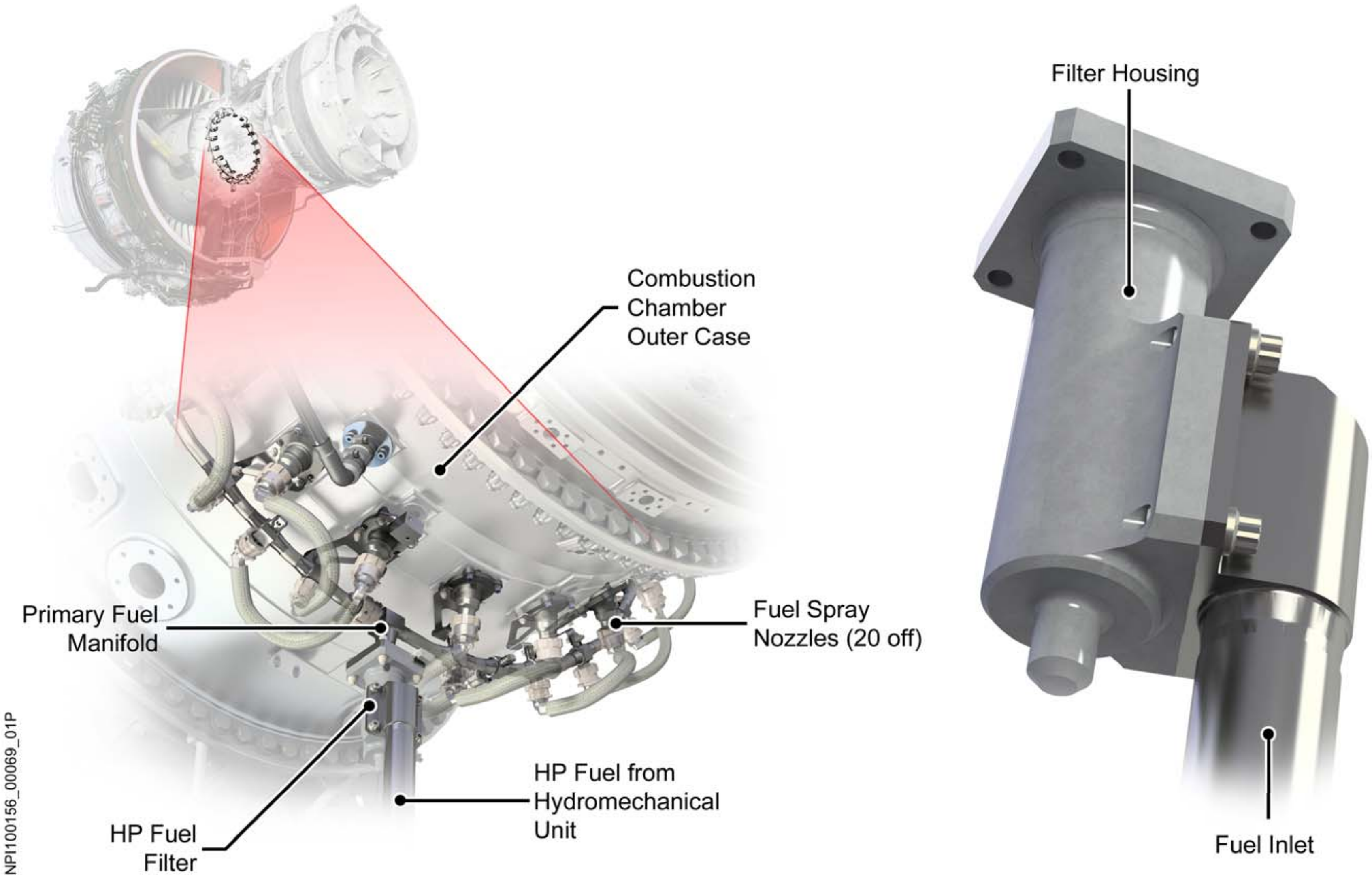
YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION

YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.

CAUTION

YOU MUST NOT LET ENGINE FUEL FALL ON THE ENGINE. REMOVE UNWANTED FUEL IMMEDIATELY WITH A CLEAN LINT-FREE CLOTH. THE FUEL CAN CAUSE DAMAGE TO THE SURFACE PROTECTION AND TO SOME PARTS.



HP FUEL FILTER INSTALLATION

Maintenance Practices.

Hydro-mechanical unit – Hydromechanical meter

(73-21-13, 01-250)

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Removal Procedure

AMP Tasks: TRENTXWB-A-73-21-13-01A01-520A-A,

TRENTXWB-A-73-21-13-01A01-520A-D

WARNING
BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING
YOU MUST NOT TOUCH HOT PARTS WITHOUT APPLICABLE GLOVES. HOT PARTS CAN CAUSE AN INJURY. IF YOU GET AN INJURY, PUT IT IN COLD WATER FOR 10 MINUTES AND GET MEDICAL AID.

WARNING
YOU MUST MAKE SURE THAT YOU CAN HOLD THE WEIGHT OF THE COMPONENT BEFORE YOU REMOVE/INSTALL IT. THE COMPONENT IS HEAVY. IF IT FALLS, IT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

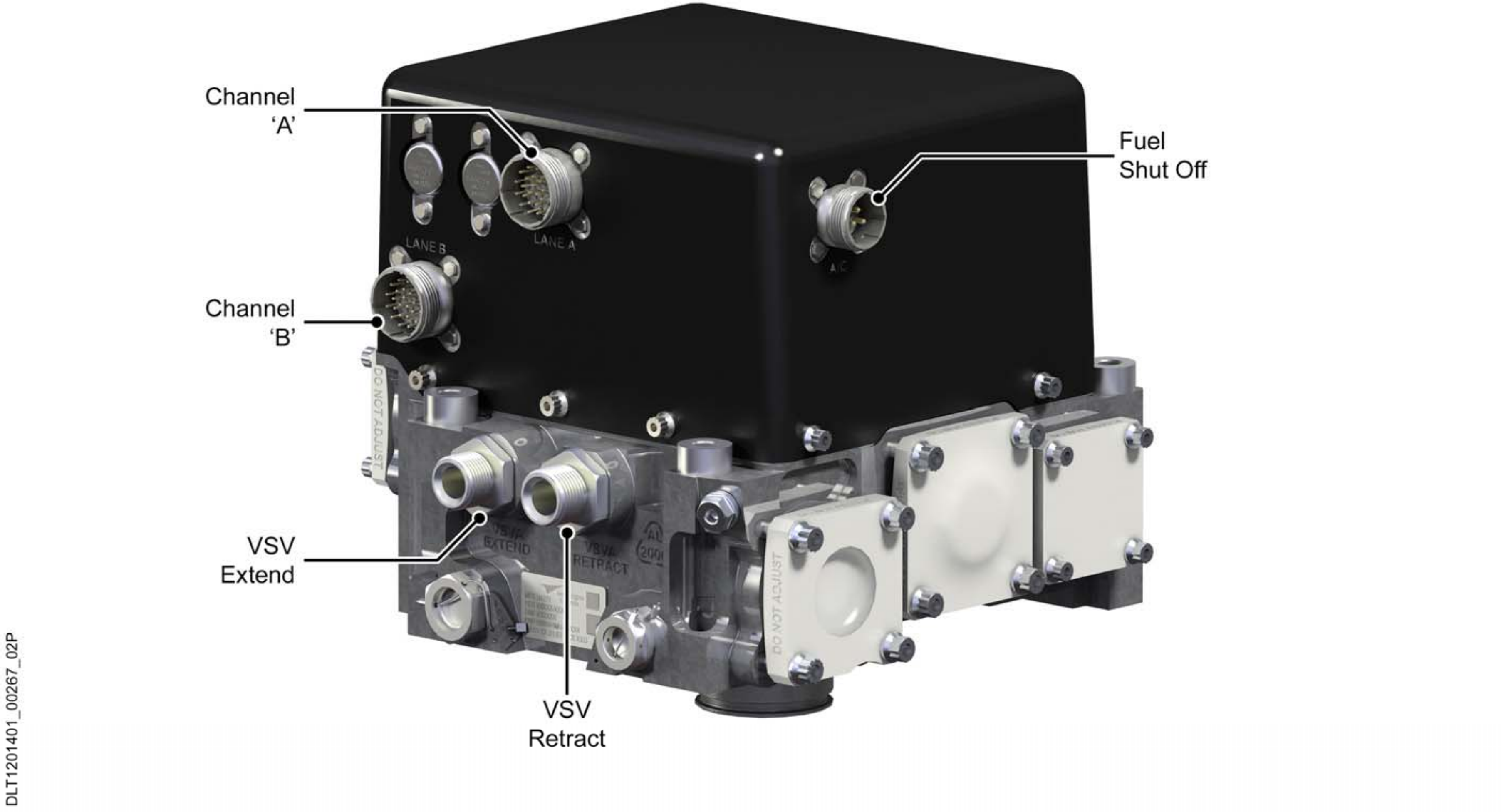
CAUTION
YOU MUST NOT BEND THE ELECTRICAL HARNESS TOO MUCH WHEN YOU DISCONNECT/CONNECT THE ELECTRICAL CONNECTORS. IF YOU DO NOT OBEY THIS INSTRUCTION, DAMAGE TO THE HARNESSES CAN OCCUR. THIS CAN CAUSE ELECTRICAL CIRCUIT DEFECTS.

CAUTION
YOU MUST NOT USE PLIERS WITH METAL JAWS TO DISCONNECT/CONNECT THE PARTS. THE METAL JAWS WILL CAUSE DAMAGE TO THE PARTS.

CAUTION
YOU MUST NOT CAUSE SCRATCHES TO THE MATING FACES OF THE HMU AND EXTERNAL GEARBOX WHEN YOU REMOVE THE HMU. IF THE MATING FACES HAVE SCRATCHES, A LEAKAGE CAN OCCUR.

Note

The HMU weighs approximately 12.5 kg (27.5 lbs).


Channel 'A'
Fuel Shut Off
LANE B
LANE A
Channel 'B'
VSV Extend
VSV Retract
DO NOT ADJUST
DLT1201401_00267_02P

HMU REMOVAL

Maintenance Practices

Hydro-mechanical unit – Hydromechanical meter

(73-21-13, 01-250)

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Install Procedure.

AMP Tasks: TRENTXWB-A-73-21-13-01A01-720A-A,

TRENTXWB-A-73-21-13-01A01-720A-D

WARNING
YOU MUST MAKE SURE THAT YOU CAN HOLD THE WEIGHT OF THE COMPONENT BEFORE YOU REMOVE/INSTALL IT. THE COMPONENT IS HEAVY. IF IT FALLS, IT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

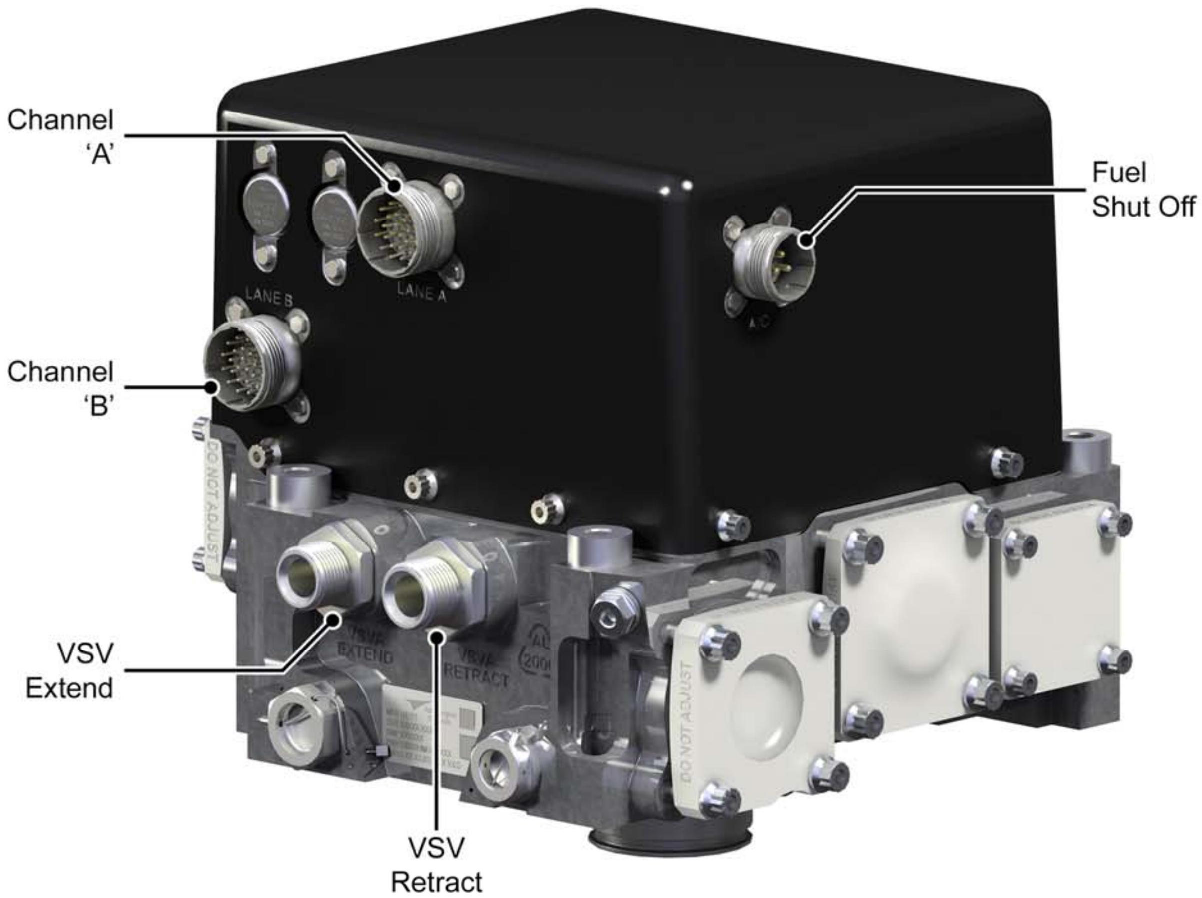
WARNING
BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

CAUTION
YOU MUST NOT CAUSE SCRATCHES TO THE MATING FACES OF THE HMU AND EXTERNAL GEARBOX WHEN YOU REMOVE THE HMU. IF THE MATING FACES HAVE SCRATCHES, A LEAKAGE CAN OCCUR.

CAUTION
YOU MUST OBEY THE INSTRUCTIONS GIVEN IN THE STANDARD PRACTISE 70-12-01 WHEN YOU REMOVE/INSTALL OR DISCONNECT/CONNECT THE TUBE(S). IF YOU DO NOT DO THIS, DAMAGE TO THE TUBE(S) CAN OCCUR.

CAUTION
YOU MUST MAKE SURE THAT THE ELECTRICAL CONNECTORS ARE CLEAN BEFORE YOU CONNECT THEM. CONTAMINATION CAN CAUSE DAMAGE TO THE EQUIPMENT.

CAUTION
YOU MUST NOT BEND THE ELECTRICAL HARNESS TOO MUCH WHEN YOU DISCONNECT/CONNECT THE ELECTRICAL CONNECTORS. IF YOU DO NOT OBEY THIS INSTRUCTION, DAMAGE TO THE HARNESSES CAN OCCUR. THIS CAN CAUSE ELECTRICAL CIRCUIT DEFECTS.



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HMU INSTALLATION

Maintenance Practices

Fuel Spray Nozzle Removal

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

Fuel Spray Nozzle Removal

AMP Task TRENTXWB-A-72-41-72-01A01-520A-A

WARNING
YOU MUST NOT GET ENGINE FUEL ON YOUR SKIN, IN YOUR EYES OR IN YOUR MOUTH. ENGINE FUEL IS AN IRRITANT. YOU MUST NOT BREATHE THE FUMES. FUEL FUMES CAN CAUSE DAMAGE TO YOUR LUNGS. PUT ON GOGGLES OR A FACE MASK. IF YOU GET ENGINE FUEL ON YOUR SKIN, FLUSH IT AWAY WITH WATER. IF YOU GET ENGINE FUEL IN YOUR EYES OR IN YOUR MOUTH, FLUSH IT AWAY WITH WATER. YOU MUST NOT CAUSE VOMITING BUT GET MEDICAL AID IMMEDIATELY.

CAUTION
YOU MUST NOT LET ENGINE FUEL FALL ON THE ENGINE. REMOVE UNWANTED FUEL IMMEDIATELY WITH A CLEAN LINT-FREE CLOTH. THE FUEL CAN CAUSE DAMAGE TO THE SURFACE PROTECTION AND TO SOME PARTS.

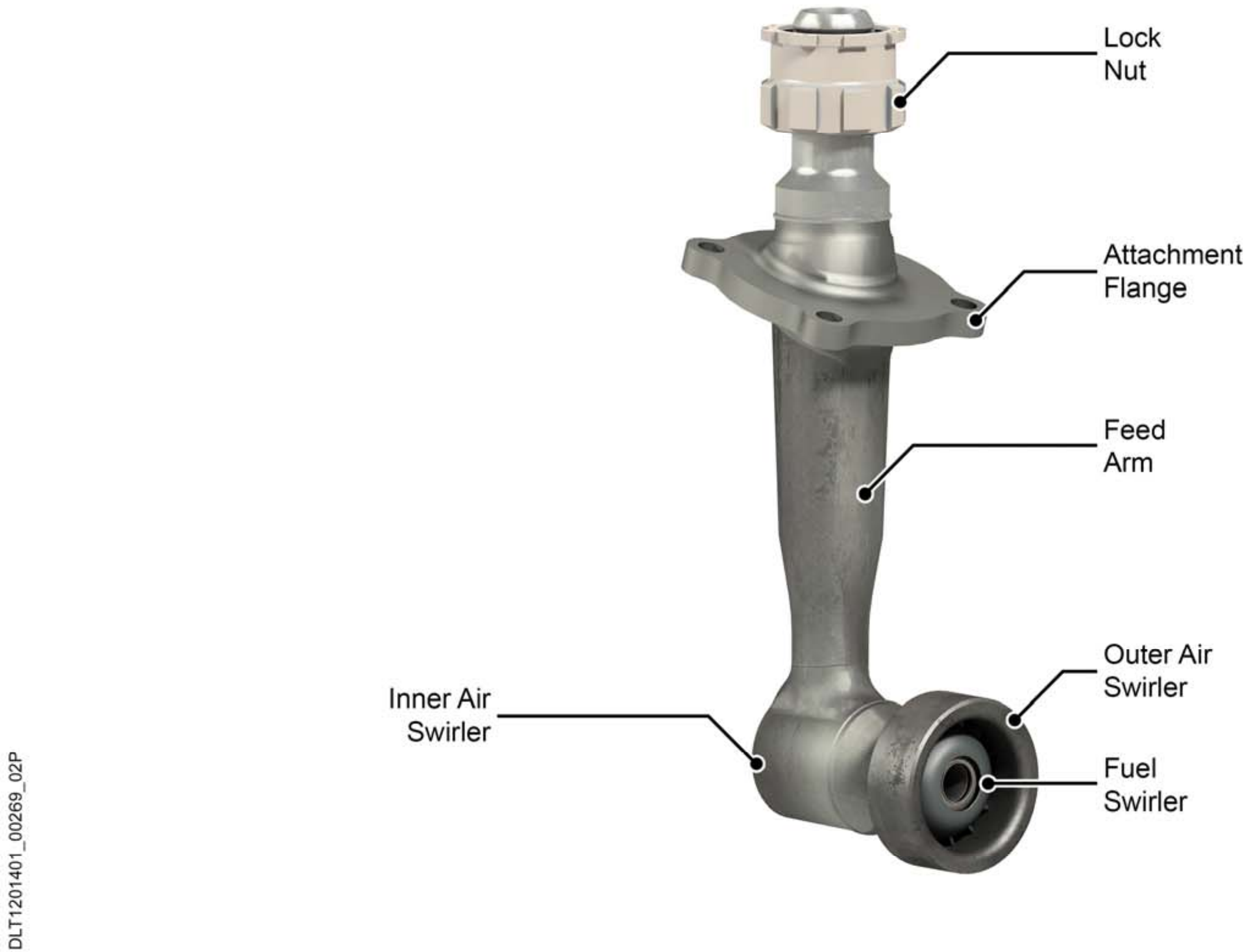
CAUTION
YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.

Note

You must retain the distributor weight assembly for installation in the replacement fuel spray nozzle.

The fuel distributor weight assembly at locations 8 and 12 are configured adjacent to the igniter plugs and are identified by a straight line on the top of the weight.

For all other locations, make sure there is a cross at the top of the fuel distributor weight assembly.



FUEL SPRAY NOZZLE REMOVAL

Maintenance Practices

Fuel Spray Nozzle Installation

The following is a brief description of these tasks. Refer to the AMP for the full description and the relevant OMSD for the correct use of any specialised tooling.

AMP Task TRENTXWB-A-72-41-72-01A01-720A-A

WARNING
BE CAREFUL WHEN YOU DO WORK ON THE ENGINE COMPONENTS AFTER THE ENGINE IS STOPPED. THE ENGINE COMPONENTS CAN STAY HOT FOR ONE HOUR.

WARNING
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CAUTION
YOU MUST NOT PUT FUEL THAT HAS BEEN DRAINED FROM THE ENGINE BACK INTO THE FUEL SYSTEM. CONTAMINATION IN THE FUEL CAN CAUSE DAMAGE TO THE ENGINE.

Note

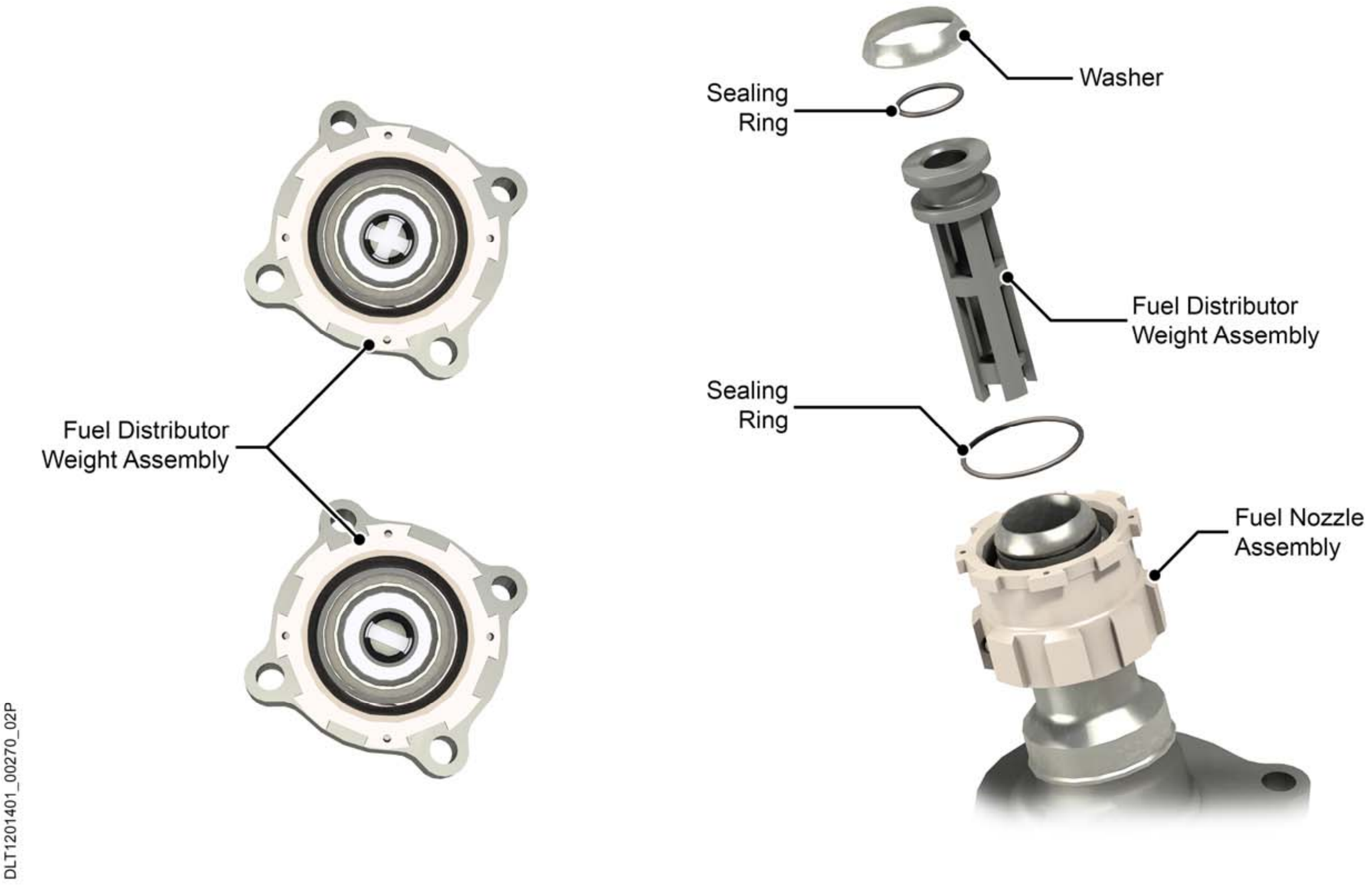
The washer CSN 72417201285 can rotate freely after installation.

Carry out a borescope inspection of the combustion inner case to ensure the Fuel nozzle assembly **CSN 7241721250** is engaged in the combustion burner seal, refer to;

TRENTXWB-A-72-41-12-03A01-300A-A

Install the main fuel hose assemblies; refer to;

TRENTXWB-A-72-41-71-03A01-720A-A



FUEL SPRAY NOZZLE INSTALLATION

On Board Maintenance – Ground or Initiated Tests

Introduction

The Initiated tests provide a means by which the maintenance personnel can conduct Engine related tests in a safe and controlled manner. The test called up is to be carried out when directed by the relevant AMP of TSM procedure.

Test Structure

The Central Maintenance Computing Function (CMCF) will generate screens to guide the operator though in the following test steps:

- Test Set-Up (indicates to the operator any necessary warnings, switch selections and instructions required to carry out the test in a safe manor).
- Test In-Progress (Indicates that the test is currently being carried out (operating) – a STOP TEST option is available if required at this point).
- Test Close-Up (switch selections and instructions required to return the system to its original condition).
- Fault Reporting (indicates Passed or Failed condition).

Below is an indication of the Fuel Metering Valve System Test, together with an explanation of the test capabilities.

Top Level Title	Function Title
Tests	System Test
	Audible Test of the Igniters
	Variable Stator Vanes Acutator Test
	Fuel Metering Valve Test
	Bleed Valve Test
	Harness Test

Performs EEC power-up checks and electrical checks of solenoid torque motor drives followed by a short period of passive monitoring Reports all detected failures.

The Fuel Metering Valve Test dry motors the engine to produce motive fuel flow that schedules the FMV to different positions to determine if it is operating correctly. The Fuel Metering Valve LVDT passes feedback of the FMV movements to the enable the EEC to compare the valves movement against a time schedule.

Reduces the fault confirmation time for failure message codes with engine harnesses allowing failures to be rapidly confirmed as faults at time of occurrence while the test is being performed.

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ENGINE GROUND INITIATED TEST

Cockpit Effects

Introduction

Below is an indication of the associated Cockpit Effect messages for this section (Fuel) together with the message type (level), an explanation of its meaning, dispatch restrictions (if any) and expected actions taken by flight or ground crew.

Note: This is for your guidance only. Due to the nature of the aircraft and engines continual development these may change and Rolls Royce cannot guarantee that they are still current and fully accurate. As such, always refer to the appropriate MMEL & AMM procedures for reference.

FUNCTION	FLIGHT DECK EFFECT	FAULT DESCRIPTION	CLASS	DISPATCH LEVEL
HP SOV	**ENG x HP FUEL VLV NOT CLOSED**	Indicates a mechanical fault of the HPSOV (i.e. stuck)	1	No Go
HP SOV	**ENG x HP FUEL VLV NOT OPEN**			
FMV Drive	**ENG x SHORT TERM MINOR FAULT**	Indicates a fault with channel ‘A’s’ FMV drive or a wiring fault to channel A of the EEC (similar fault indications and dispatch conditions for channel B of the EEC)	1	Go (STD)
	ENG x CTL SYS FAULT			No Go
	ENG x FADEC SYS FAULT			
FMV Pos	**ENG x SHORT TERM MINOR FAULT**	Indicates a fault with channel ‘A’s’ Fuel Metering Valve position sensor or a wiring fault to channel A of the EEC (similar fault indications and dispatch conditions for channel B of the EEC)	1	Go (STD)
FMV Pos	**ENG x CTL SYS FAULT**			No Go
HMU Calibration	**ENG x LONG TERM MINOR FAULT**	Indicates calibration errors within the FMV or a Fuel Flow meter fault	1	Go (LTD)
HP SOV Pos	**ENG x LONG TERM MINOR FAULT**	Indicates a fault with channel ‘A’s’ HPSOV position sensor or a wiring fault to channel A of the EEC (similar fault indications and dispatch conditions for channel B of the EEC)	1	Go (LTD)
	ENG x FADEC SYS FAULT			No Go

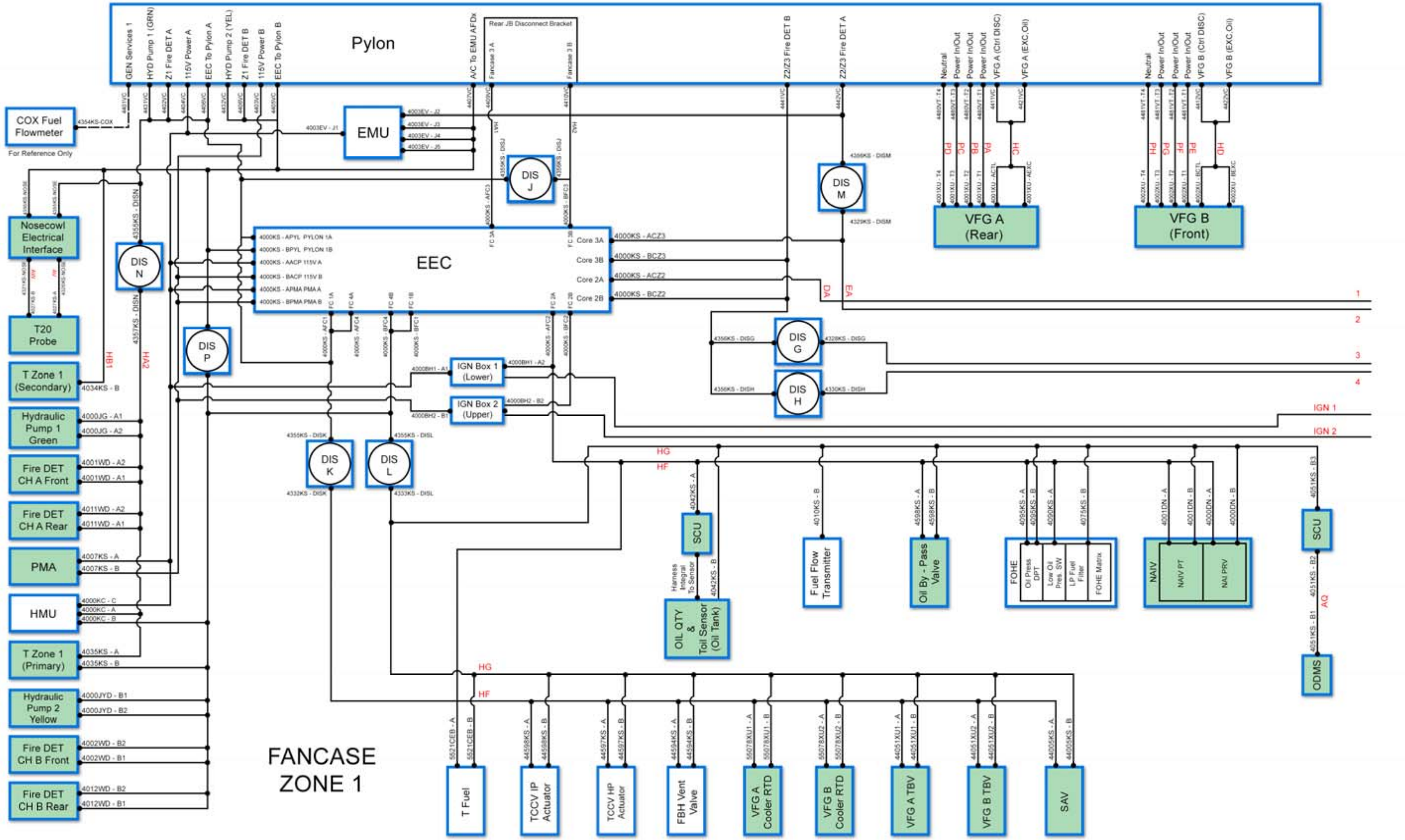
FUNCTION	COCKPIT EFFECT	FAULT DESCRIPTION	CLASS	DISPATCH LEVEL
Fuel Flow	**ENG x FUEL FLOW DEGRADED** **Fuel Flow displayed with amber dashes across the least significant three digits**	Indicates a wiring failure between the EEC and fuel flow transmitter or a mechanical failure of the fuel flow transmitter resulting in erratic or total loss of flow fate output to the EEC	1	Go (STD) – 10 days
LP Fuel Filter DP	**ENG x SHORT TERM MINOR FAULT**	Indicates a wiring or sensor failure of the LP fuel filter differential pressure transducer	1	Go (STD)
LP Fuel Filter	**ENG x FUEL FILTER PARTLY CLOGGED**	Indicates impending fuel filter blockage	1	Go (STD) – 50 hours
LP Fuel Filter	**ENG x FUEL FILTER CLOGGED**			No Go
TFuel	**ENG x LONG TERM MINOR FAULT**	Indicates a fault resulting in cross check failure with the fuel temperature sensor	1	Go (LTD)
TFuel	**None**	Indicates a fault with channel ‘A’s’ fuel temperature sensor or a wiring fault to channel A of the EEC (similar fault indications and dispatch conditions for channel B of the EEC)	1	Go (UNL)
	ENG x LONG TERM MINOR FAULT			Go (LTD)

<table>
<tr><th>FUNCTION</th><th>COCKPIT EFFECT</th><th>FAULT DESCRIPTION</th><th>CLASS</th><th>DISPATCH LEVEL</th></tr>
<tr><td>FOHE</td><td></td><td></td><td></td><td></td></tr>
<tr><td>FMV</td><td>ENG x CTL SYS FAULT</td><td>Indicates a mechanical jam of the FMV resulting in inability of the valve to track</td><td>1</td><td>No Go</td></tr>
<tr><td rowspan="2">FMV Pos</td><td>ENG x SHORT TERM MINOR FAULT</td><td rowspan="2">Indicates a fault with channel ‘A’s’ Fuel Metering Valve position sensor or a wiring fault to channel A of the EEC (cross check failure with ‘A’ deemed to have failed
(similar fault indications and dispatch conditions for channel B of the EEC)</td><td rowspan="2">1</td><td>Go (STD)</td></tr>
<tr><td>ENG x SHORT TERM MINOR FAULT</td><td>Go (STD)</td></tr>
<tr><td>HP SOV Pos</td><td>ENG x LONG TERM MINOR FAULT</td><td rowspan="2">Indicates a fault with channel ‘A’s’ HPSOV position sensor or a wiring fault to channel A of the EEC (cross check failure with ‘A’ not matching demand)
(similar fault indications and dispatch conditions for channel B of the EEC)</td><td rowspan="2">1</td><td>Go (LTD)</td></tr>
<tr><td>HP SOV Pos</td><td>ENG x LONG TERM MINOR FAULT</td><td>Go (LTD)</td></tr>
<tr><td rowspan="3">HMU Calibration</td><td>None</td><td rowspan="3">Indicates a failure of the HMU smart calibration data or link to channel A of the EEC
(similar fault indications and dispatch conditions for channel B of the EEC)</td><td>1</td><td>Go (UNL)</td></tr>
<tr><td>ENG x SHORT TERM MINOR FAULT</td><td rowspan="2">1</td><td>Go (STD)</td></tr>
<tr><td>ENG x CTL SYS FAULT</td><td>No Go</td></tr>
</table>

Fuel System Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMM or TSM electronic documentation.



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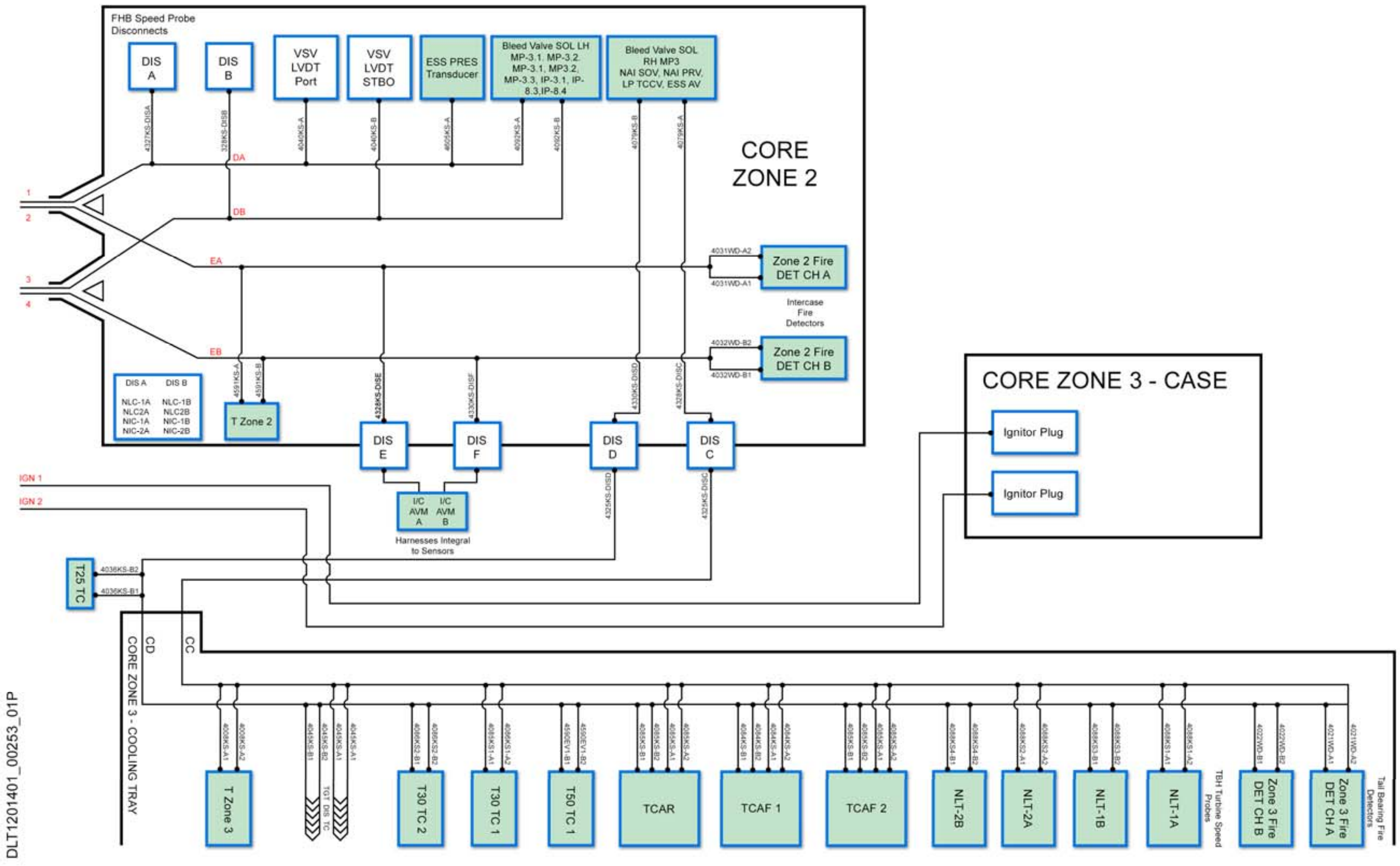
ZONE 1 WIRING DIAGRAM

Fuel System Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc) and for further details, then always refer to the actual wiring diagrams as given within the AMM or FIM electronic documentation.



ZONE 2 & 3 WIRING DIAGRAM

Section 8 - Engine Fuel System

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Control and Fuel System as fitted to the Trent XWB engine.
- Locate and identify the LRU's that form the Engine Control and Fuel System of the Trent XWB engine.
- Describe the purpose and operation of the LRU's that form the Engine Control and Fuel System of the Trent XWB engine.
- State the WARNINGS & CAUTIONS associated with the Engine Control and Fuel System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Control and Fuel System interface with other engine and aircraft systems.

End of Fuel Section

Section 9 - Airflow Control

Section 9 - Airflow Control

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the engine Airflow Control System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Airflow Control System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs of the engine Airflow Control System as fitted to the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the engine Airflow Control System as fitted to the Trent XWB engine.
- Describe how the Trent XWB engine Airflow Control System interfaces with other engine and aircraft systems.

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Airflow Control System

Location

The airflow control system components are located on the LP compressor case and engine core.

Purpose

The purpose of the airflow control system is to provide a stable airflow through the engine compressors at all operating conditions, protecting against stall and surge and maintaining a high degree of efficiency.

Description

The airflow control system is controlled by the EEC using the following components:

- One stage of variable inlet guide vanes (VIGVs).
- Two stages of variable stator vanes (VSVs).
- Two VSV / VIGV Actuators
- Three IP8 bleed valves (Handling Bleeds).
- Three HP3 bleed valves (Handling Bleeds).
- Two bleed valve solenoid banks.
- The HMU

HMU

The HMU controls the position of the VIGV/VSV system in response to signals through electrical harnesses from the EEC.

VIGV / VSV System

The VIGVs and VSVs control the angle of the air supplied to the first three rotor stages of the IP compressor. The angle of the VIGVs and VSVs, normally closed at low speed and open at high speeds, vary in response to acceleration and deceleration commands to optomise performance and prevent stall and surge conditions occurring.

IP and HP Compressor Bleed Valves

The Three IP8 and HP3 compressor bleed valves exhaust compressor air into the bypass duct when commanded by the EEC through their individual solenoids.

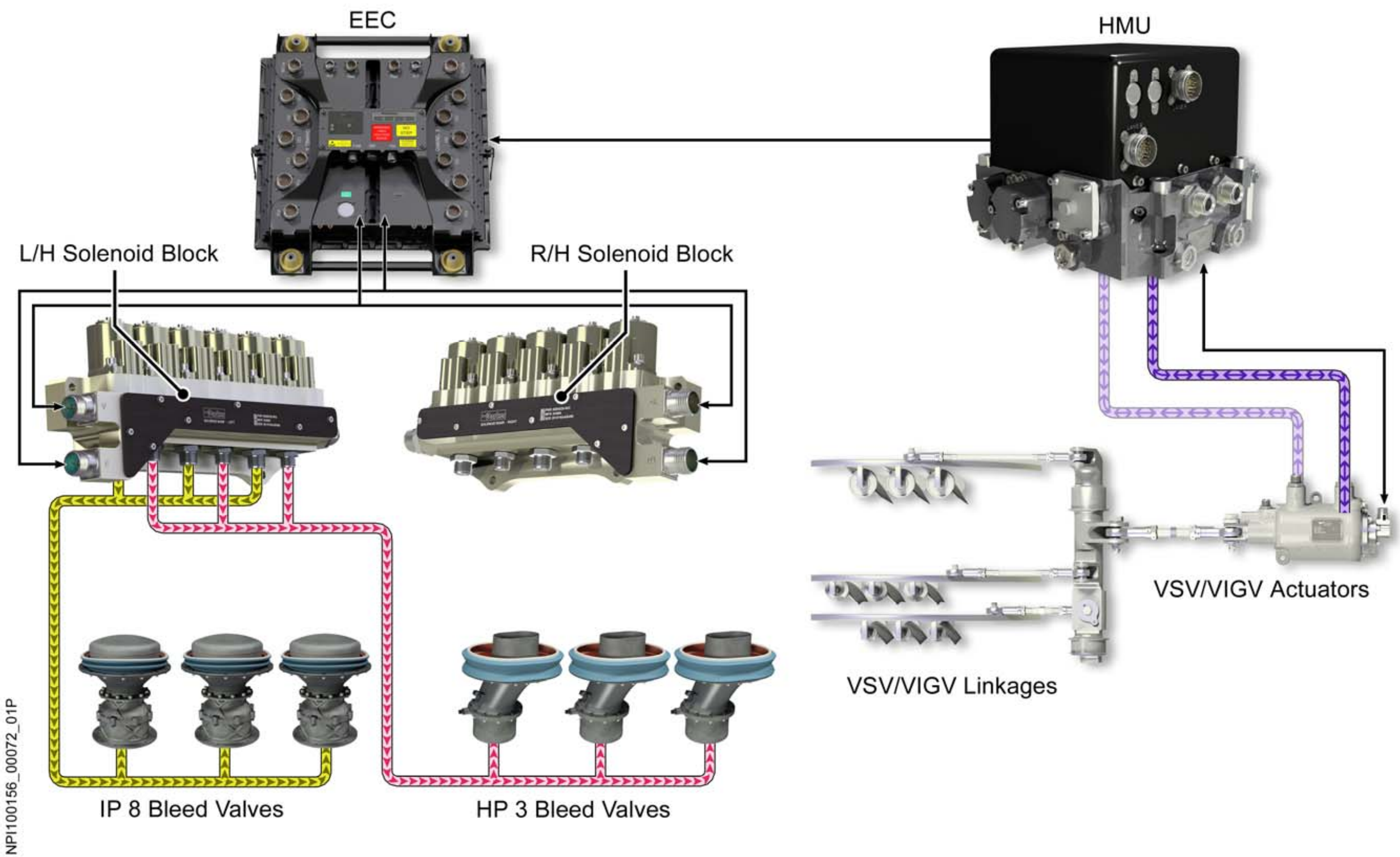
Solenoid Banks

There are two solenoid banks.

One installed on the left side of the Intermediate case and the other is installed on the right side of the Intermediate case.

The six solenoids in the left bank operate the IP 8 / HP 3 handling bleed valves.

The five solenoids in the right bank operate the, LPTCCV, ESS anti-ice valve, and the nacelle anti-ice PRSOV. There is one spare solenoid on this bank due to the deletion of a now redundant system.



AIRFLOW CONTROL SYSTEM

Variable Inlet Guide Vane (VIGV) / Variable Stator Vane (VSV) Control System

Location

The VIGVs are located immediately behind the Engine Section Stators (ESS). The VSVs are located after the first and second stages of the IP Compressor rotors.

Purpose

The Variable Stator Vane Actuator System changes the angular position of the VSVs and VIGVs.

This controls the air flow supplied to the first three rotor stages of the IP compressor.

The position of the VIGVs and VSVs is usually open at high speeds and closed at low speeds.

The angular position will also change with acceleration and deceleration to makes sure the performance is efficient and to prevent stall and surge conditions.

Description

The VIGV/VSV system consists of the following units:

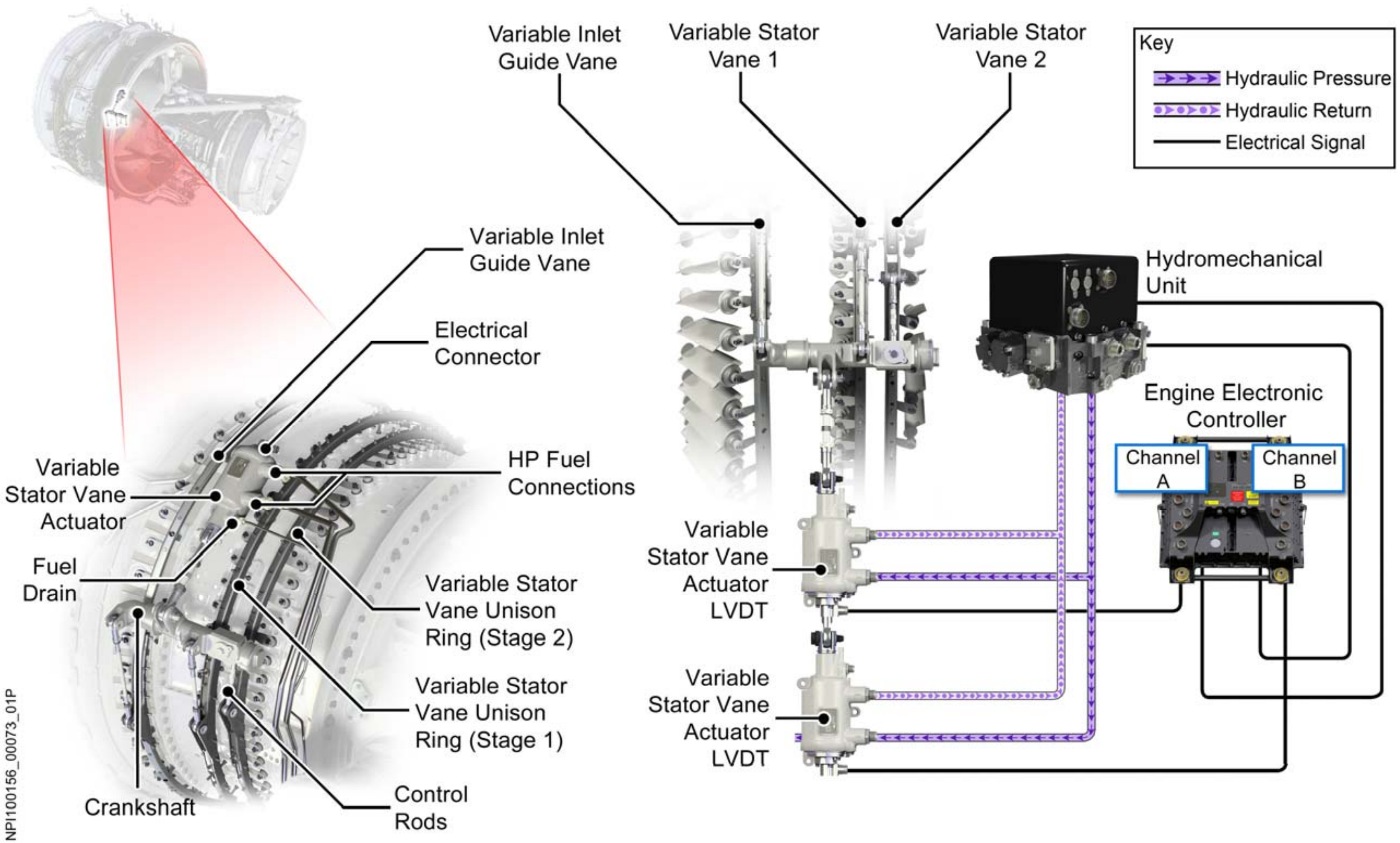
- The Variable Inlet Guide Vanes (VIGVs)
- The Variable Stator Vanes (VSVs)
- The Variable Stator Vane X2 Actuators
- The X2 Crankshaft and Control rod assemblies
- The X3 Unison rings made of x 6 halves

The VIGV / VSV system is controlled by the EEC via a dual channel two stage VSV Servo valve inside the HMU.

Fuel is used as the hydraulic operating medium (Muscle force) and a single channel Linear Variable Differential Transducer (LVDT) inside each actuator supplies actuator ram position to the EEC.

The left actuator LVDT provides a signal to Channel A, and the right actuator LVDT provides a signal to Channel B.

The EEC channel in control only uses the positional feedback from its own LVDT unless that signal is lost, in which case, it will then use the input signal from the other channel.



VARIABLE INLET GUIDE VANE/STATOR VANE ACTUATOR SYSTEM

Variable Inlet Guide Vane (VIGV) / Variable Stator Vane (VSV) Control System Continued

Variable Stator Vanes

The VSV mechanisms are installed on the IP compressor case this includes a crankshaft, control rods and unison rings. The VSV actuators are connected to the VSV mechanism. This mechanism changes the linear movement of the actuator to angular movement of the VSVs through their specified range. The VSV actuators are hydraulically extended and retracted with HP fuel supplied from the Hydro-Mechanical Unit (HMU).

Variable Inlet Guide Vanes

The VIGV mechanisms are installed on the IP compressor case this includes a crankshaft, control rods and unison rings. The VSV actuators are connected to the VIGV mechanism. This mechanism changes the linear movement of the actuator to angular movement of the VIGVs through their specified range. The VSV actuators are hydraulically extended and retracted with HP fuel supplied from the Hydro-Mechanical Unit (HMU).

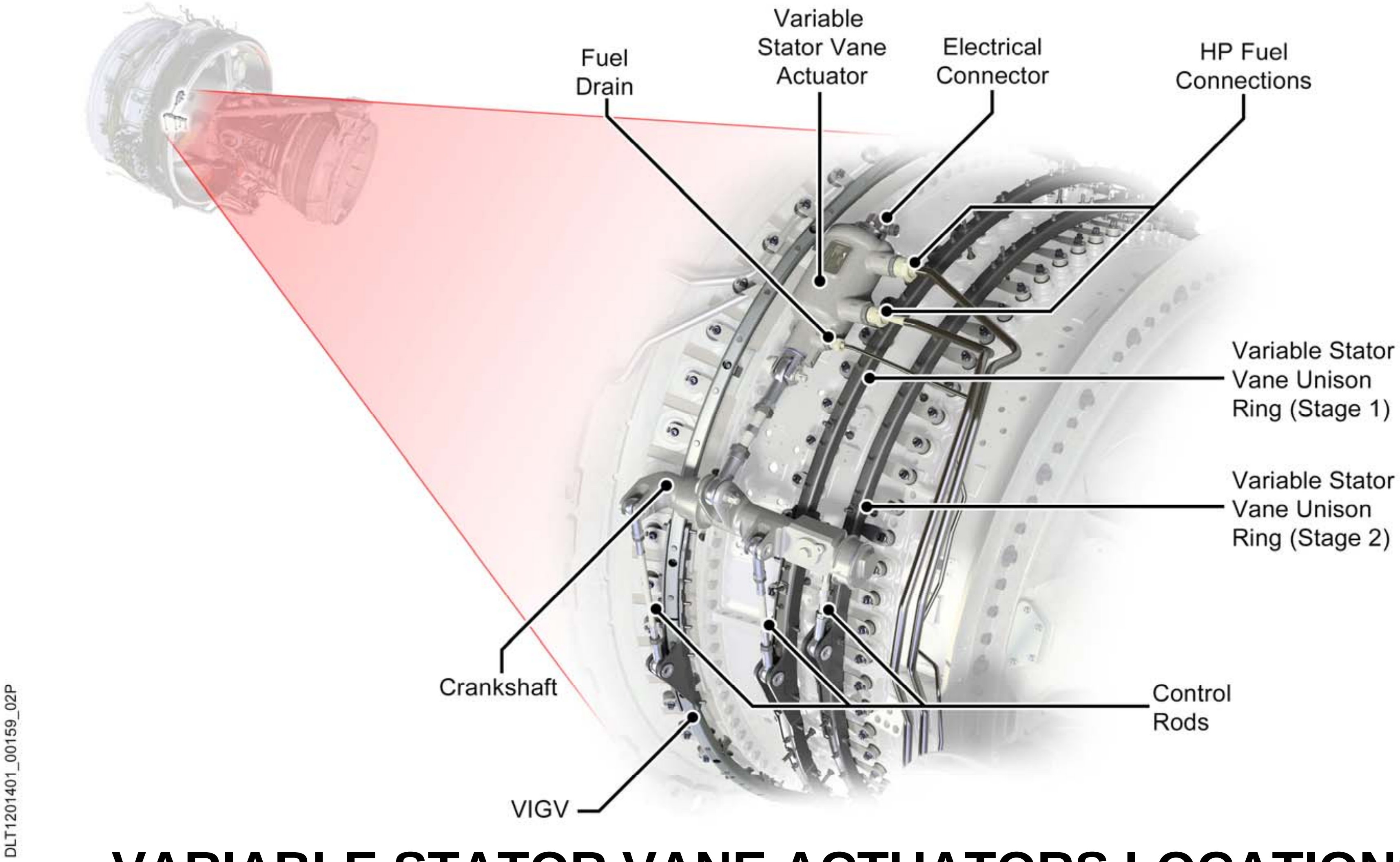
VIGV / VSV Actuators

The two combined VIGV / VSV actuators are installed around IP compressor case. Each actuator has two HP fuel connections, one fuel drain connection and one electrical connection. The HP fuel connections supply fuel to extend and retract the actuator rams. The fuel drain connection will give an indication of an internal failure of the actuator. The electrical connection connects the actuator to its related EEC channel. A Linear Variable Differential Transducer (LVDT) installed in the actuator monitors the position of the hydraulically operated ram and transmits electrical signals to the EEC. The HMU controls the direction of the fuel flow to the HP fuel connections on the actuators to extend or retract the ram. The HMU is controlled by electronic signals from the EEC.

Location

The locations of the VIGV / VSV actuating system components are as follows:

- The VIGV / VSV actuators are located 180 degrees apart at the 5 o’clock and 10 o’clock positions on the IP compressor case.
- The VIGV / VSV mechanism is located around the IP compressor.



VARIABLE STATOR VANE ACTUATORS LOCATIONS

Variable Inlet Guide Vane (VIGV) / Variable Stator Vane (VSV) Control System Continued

Normal Operation

Variable Stator Vane Actuator System

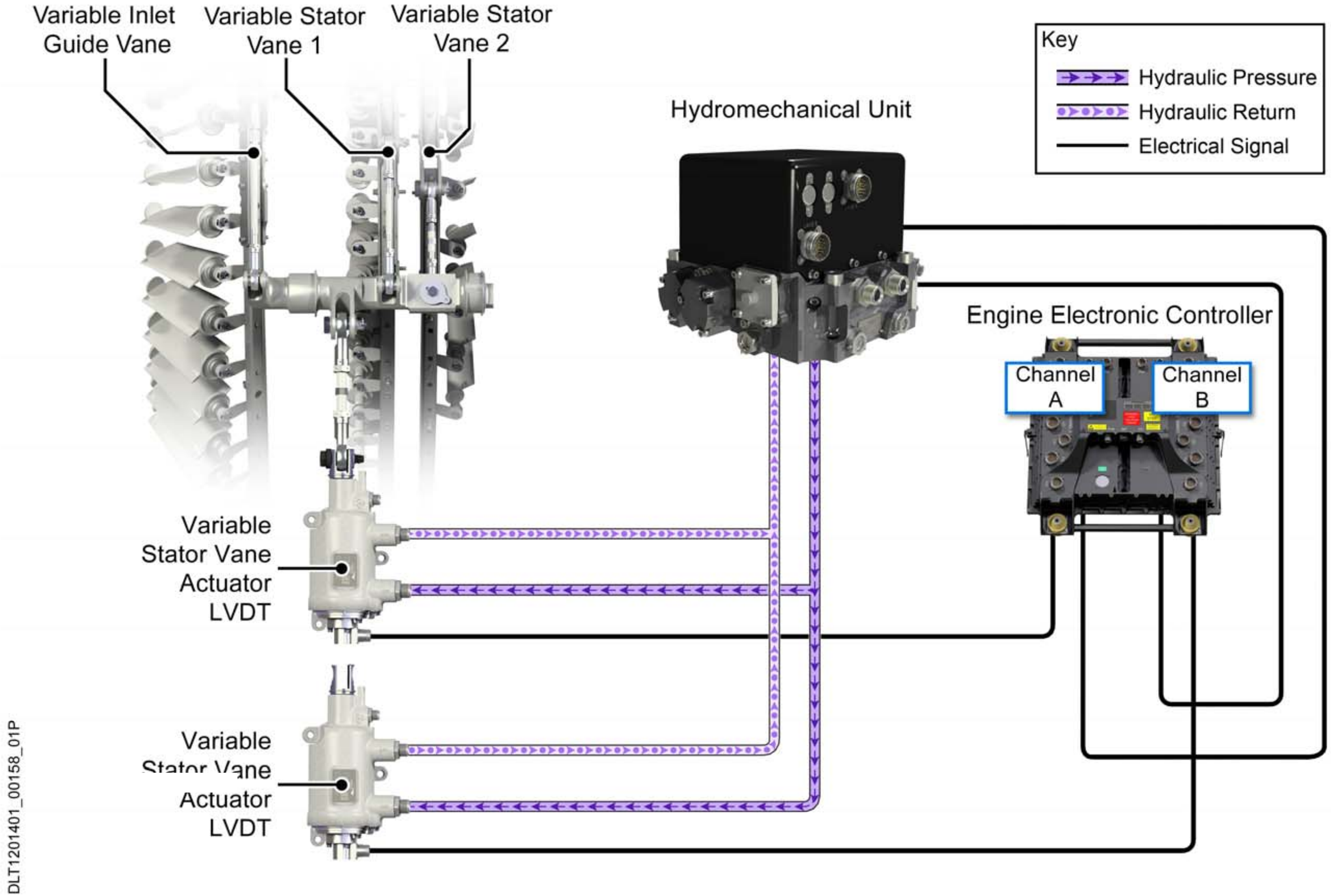
During engine start the VIGVs and VSVs are held in their closed position. They are held in the closed position until there is sufficient HP servo fuel pressure to operate the actuators and open the VIGV / VSV.

In acceleration or as the engine speed increases the VIGV / VSVs will move to their fully open position. The EEC uses N2 (IP compressor speed) and IP compressor inlet temperature to control the angular position of the VIGV / VSV.

Failsafe position

On system failure the VSV mechanism will go to the open position (High speed position)

This will prevent the shaft over speeding by loading the compressor with air.



VIGV / VSV CONTROL SYSTEM

Compressor Handing Bleed Valve System

Location

- The IPBV's are positioned on the IP case at the following locations ALF the 3, 7, & the 9 o'clock positions.
- The HPBV's are positioned on the HP case at the following locations ALF the 10, 4, & 2 o'clock positions.

Purpose

The purpose of the IP8 and HP3 bleed valves controls the flow of air through the compressors during engine operations.

Description

The Air Bleed System consists of the following components.

- The Left Hand Solenoid Block
- The IP8 Bleed Valves
- The HP3 Bleed Valves.

At engine start the bleed valves are open due to spring force; this allows air to be bled off to reduce the loads on the compressor during the start sequence.

At the lower engine speeds the bleed valves are held open by spring force plus HP3 servo air to bleed some of the compressor airflow into the bypass duct to prevent stall or surge.

The bleed valves are closed at higher engine speeds to allow full airflow through the IP and HP compressors.

The bleed valves are pneumatically opened and closed with HP3 servo air supplied from the left solenoid block.

The Left Hand bleed valve solenoid block is controlled by signals from the Engine Electronic Controller (EEC).

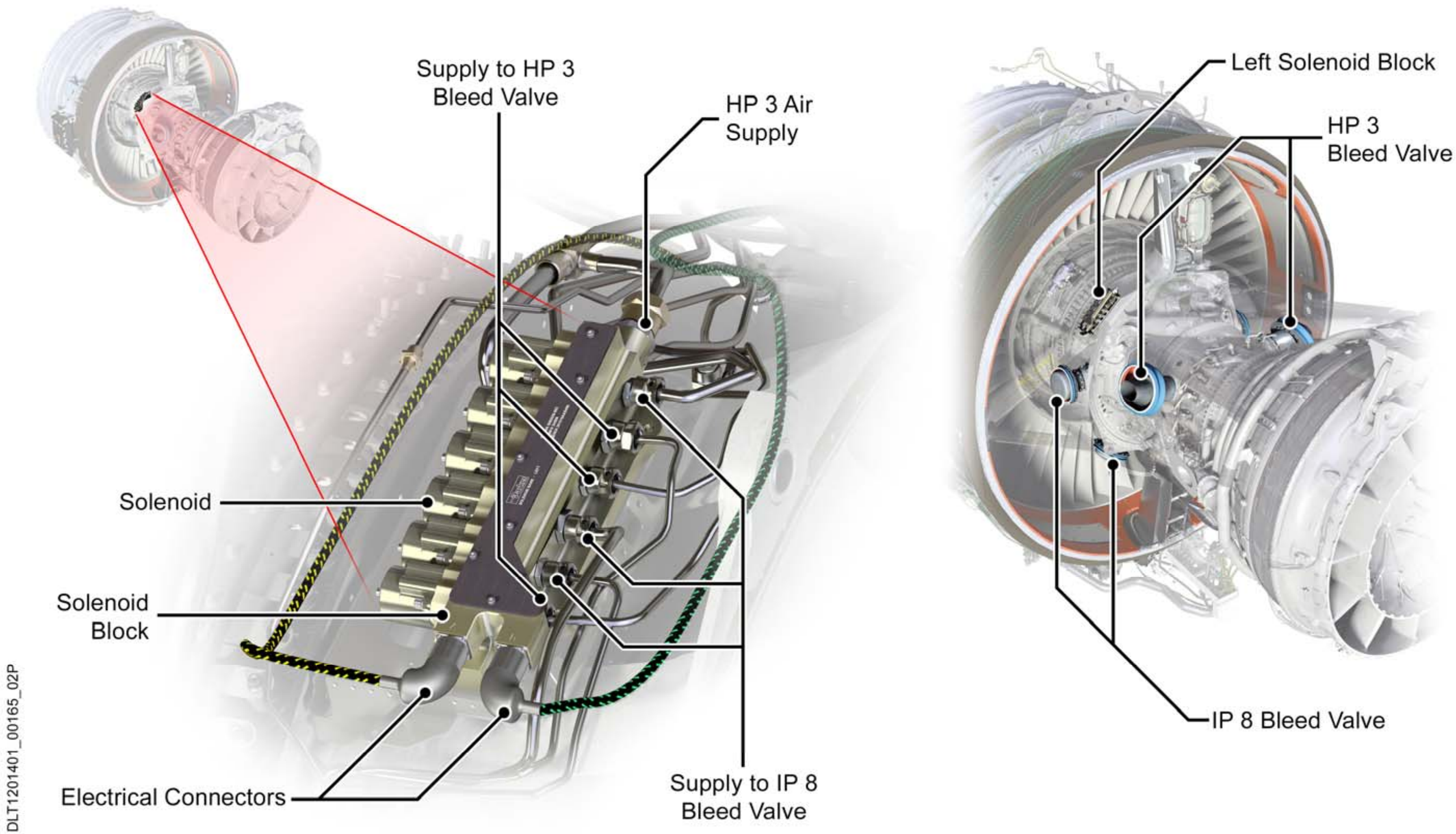
There is no bleed valve positional feedback to the EEC.

The EEC uses references from N2 (IP compressor speed) and IP compressor inlet temperature to control the IP bleed valve positions.

The EEC uses references from N3 (HP compressor speed) and IP compressor inlet temperature to control the HP bleed valve positions.

Failsafe Control

If a failure of the electrical signals to the solenoid occurs, the system is designed for the handling bleed valves to automatically close (high speed position). This failure mode makes sure that the engine internal air pressure distribution does not adversely affect turbine cooling.



COMPRESSOR BLEED VALVE SYSTEM

Compressor Handing Bleed Valve System continued

IP8 Bleed Valve

Location

There are three IP8 bleed valves installed on the Intermediate case.

Description

HP3 servo air from the left bleed valve solenoid block is delivered to each bleed valve when commanded by the EEC to move the valve to the open position. A diffuser is attached to each bleed valve outlet which reduces the hot airflow force when mixing with bypass airflow.

Valve Operation

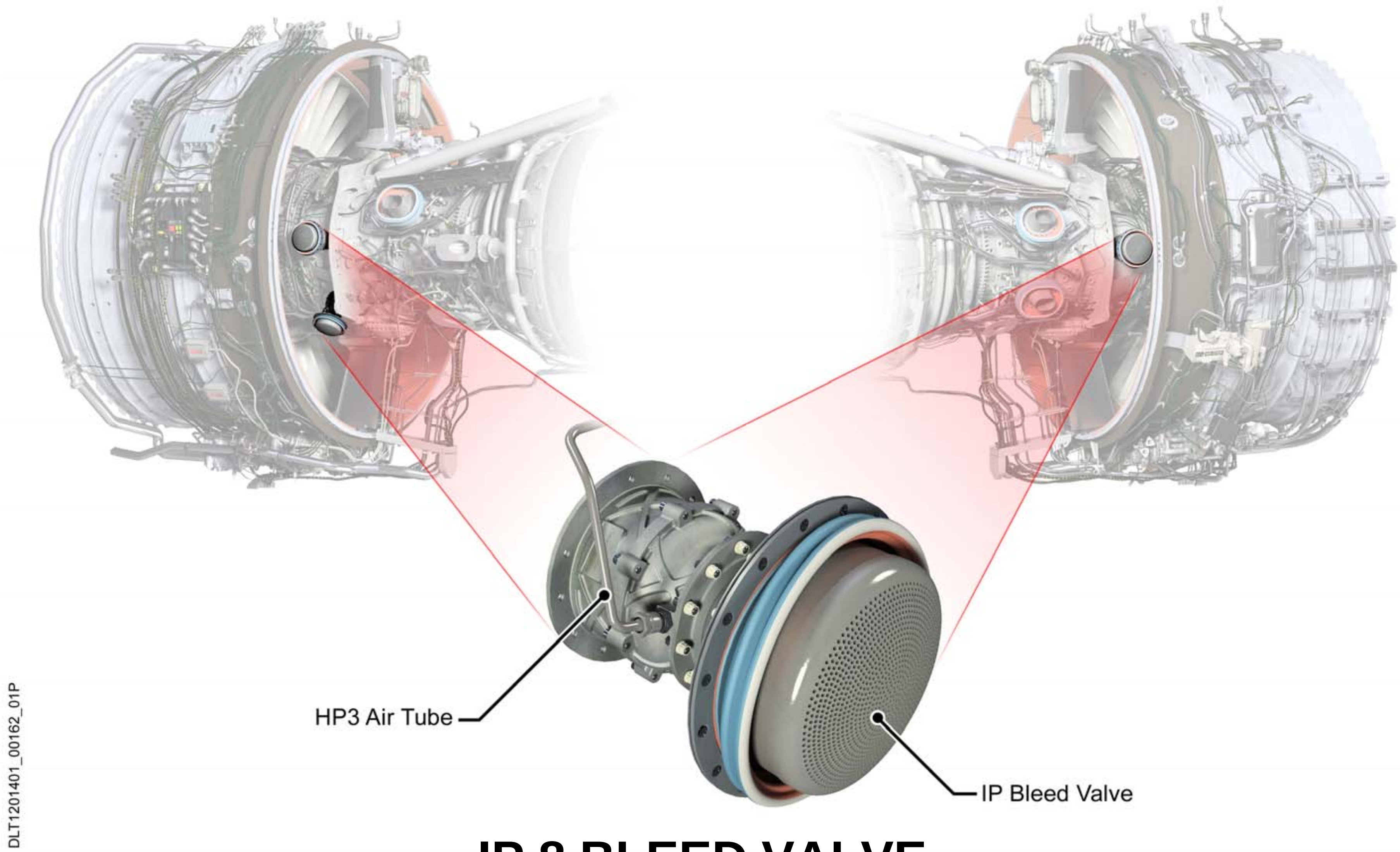
In the static position the bleed valves are spring loaded to the open position this gives the correct airflow through the IP compressor for engine start.

During the starting sequence the IP8 compressor air acts on the piston overcoming the spring force; this causes the valve to move to the closed position.

To open the valves the EEC sends an electrical signal to a solenoid in the left solenoid block. This energises the solenoid and HP3 air is then supplied to the bleed valve. The HP3 air and spring pressure moves the valve towards the fully open position.

To close the valves the solenoid is de-energised the HP3 air is vented to atmosphere. The bleed valve will then move towards the closed position controlled by the spring and IP8 compressor air inlet pressure.

The EEC schedules the IP handling bleed valves according to a measure of the engine IP compressor inlet air mass flow; using IP shaft speed, LP shaft speed and T20 signals.



IP 8 BLEED VALVE

Compressor Handing Bleed Valve System continued

HP3 bleed valves

Location

There are three HP3 bleed valves installed on the HP compressor case.

Description

HP3 servo air from the left bleed valve solenoid block is delivered to each bleed valve when commanded by the EEC to move the valve to the open position. A diffuser is attached to each bleed valve outlet which reduces the hot airflow force when mixing with bypass airflow.

Valve Operation

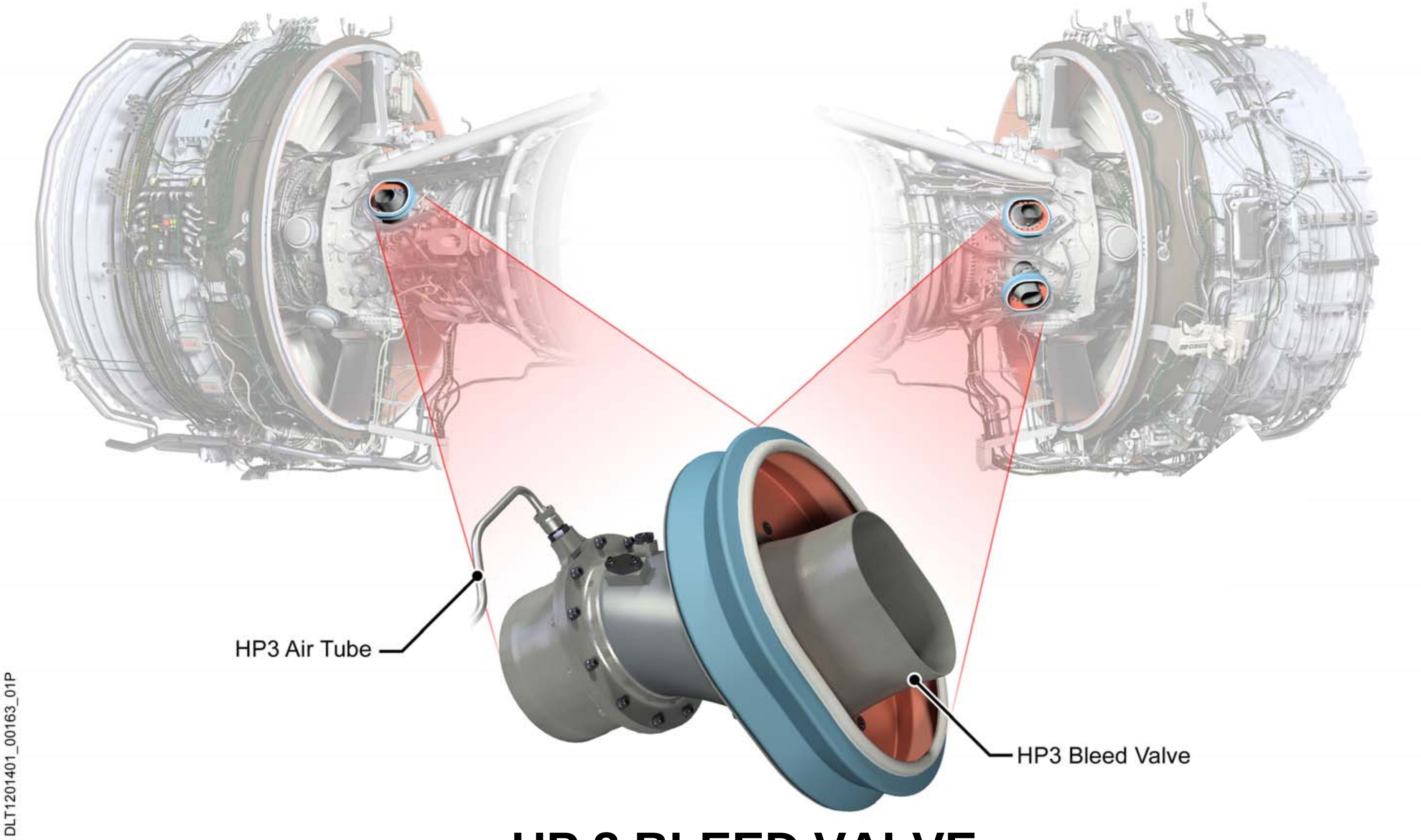
In the static position the bleed valves are spring loaded to the open position this gives the correct airflow through the HP compressor for engine start.

During the starting sequence the HP3 compressor air acts on the piston overcoming the spring force; this causes the valve to move to the closed position.

To open the valves the EEC sends an electrical signal to a solenoid in the left solenoid block. This energises the solenoid and HP3 air is then supplied to the bleed valve. The HP3 air and spring pressure moves the valve towards the fully open position.

To close the valves the solenoid is de-energised the HP3 air is vented to atmosphere. The bleed valve will then move towards the closed position controlled by the spring and HP3 compressor air inlet pressure.

The EEC schedules the HP handling bleed valves according to a measure of the engine HP compressor inlet air mass flow; using HP shaft speed, LP shaft speed and T20 signals.

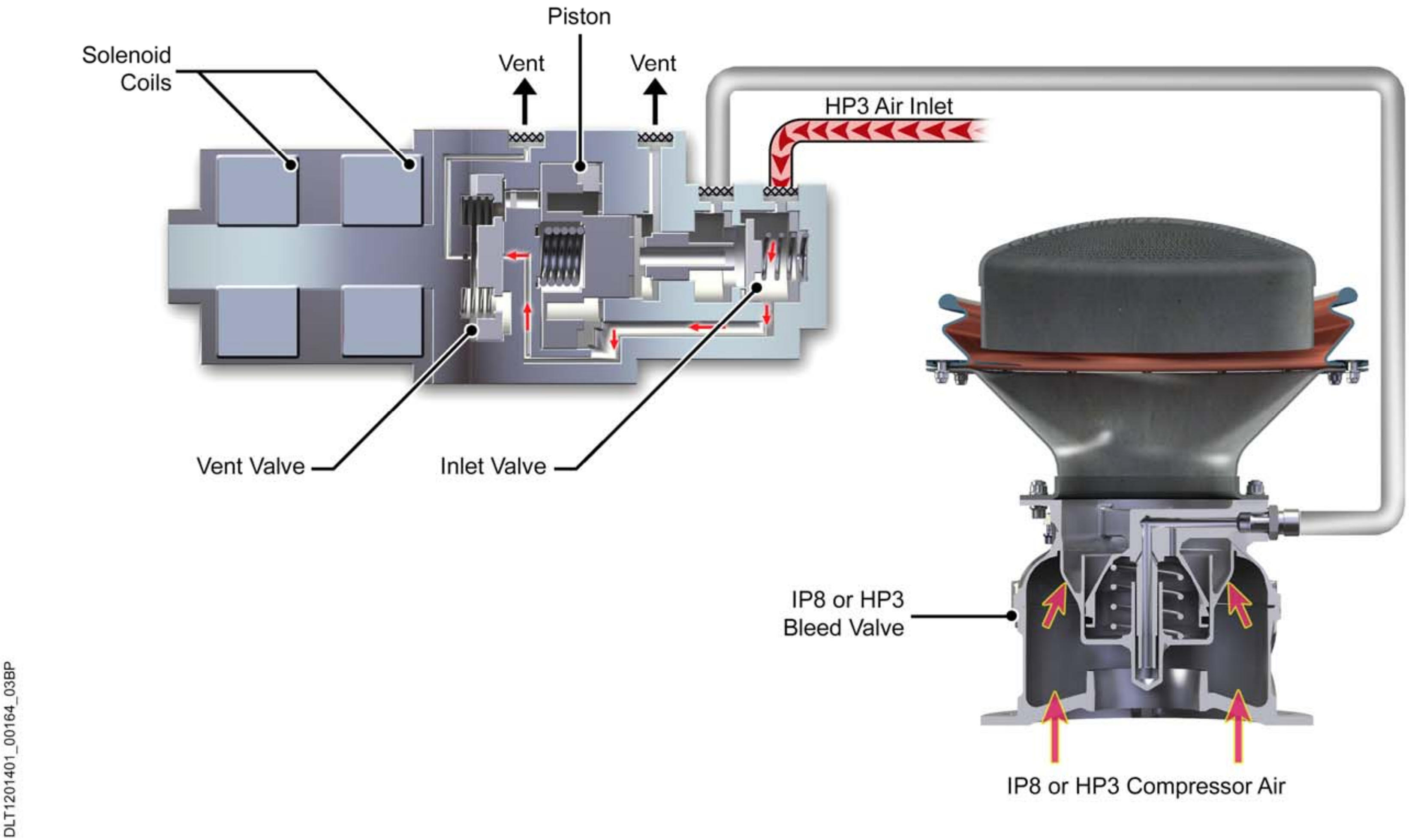


HP 3 BLEED VALVE

Bleed Valve and Solenoid Operation

Closed position

When the solenoid valve coils are de-energised, the pilot valve is positioned to allow HP3 Servo pressure through to vent. The compressor air acting against the bleed valve spring pressure will then close the valve.



BLEED VALVE OPERATION CLOSED

Bleed Valve and Solenoid Operation

Open position

To open the bleed valve the solenoid is energised moving the pilot valve closing the vent allowing HP3 air to the spring chamber of the Bleed Valve.

This force assists the valve spring acting against the compressor delivery air and so holding the valve to the open position.

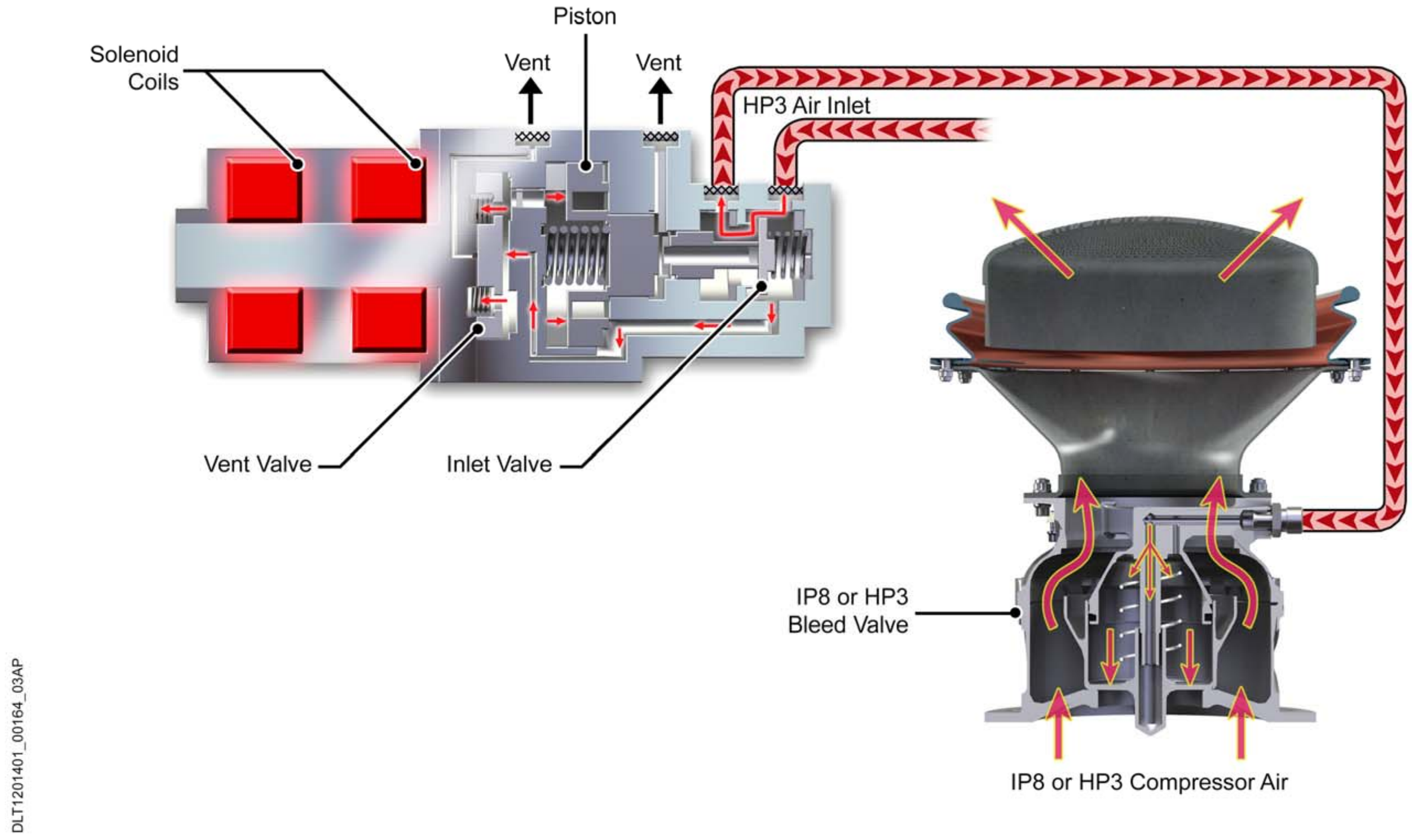
Fault Annunciation

The EEC can carry out continuity checks between the EEC and the bleed valve controllers and will set a fault message for failure of continuity. However, there is no feedback to the EEC to confirm that the bleed valve has mechanically operated correctly. If a bleed valve is not operating it will show itself by either of the following:

Valve opens when it should be closed – this will bleed air from the compressor at the higher rpm range and will show an increase in TGT. This may be observed by the aircrew, but will certainly show itself on condition monitoring as a step change.

Valve closed when it should be open – this is likely to show itself during starting with a tendency to cause hot/stagnated starts.

A bleed valve scheduling test can be carried out on the ground, with the engine running at idle. The EEC commands each of the bleed valves open & closed and reports any faults by monitoring changes in engine conditions.



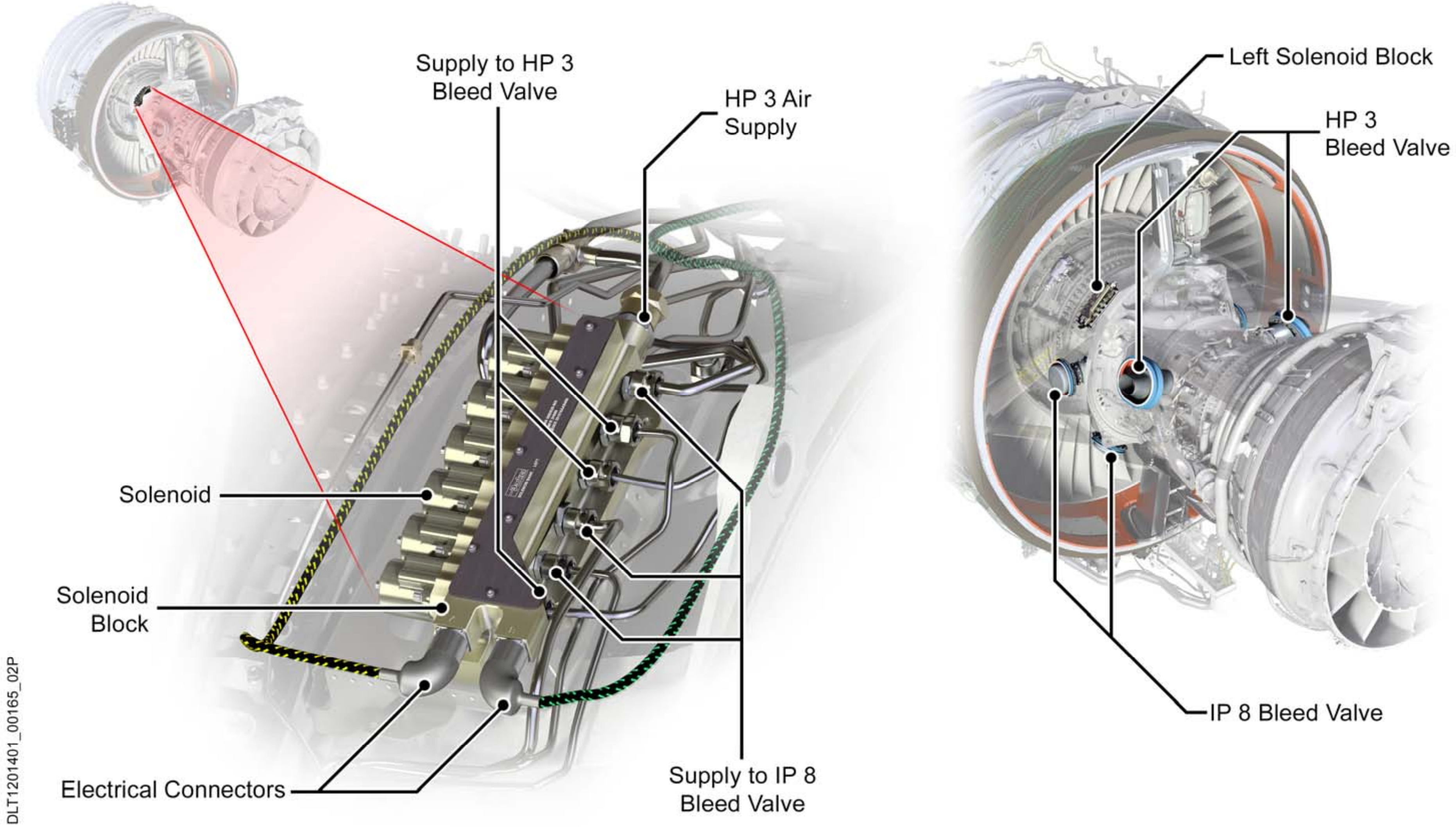
BLEED VALVE OPERATION OPEN

Left solenoid block

The left bleed solenoid block contains six solenoid valves that operate the three IP8 bleed valves and the three HP3 bleed valves.

Each of the solenoid valves can be independently operated with electronic signals from the EEC.

The bleed valves are pneumatically opened and closed, as air is supplied or released to atmosphere by the solenoid valves.

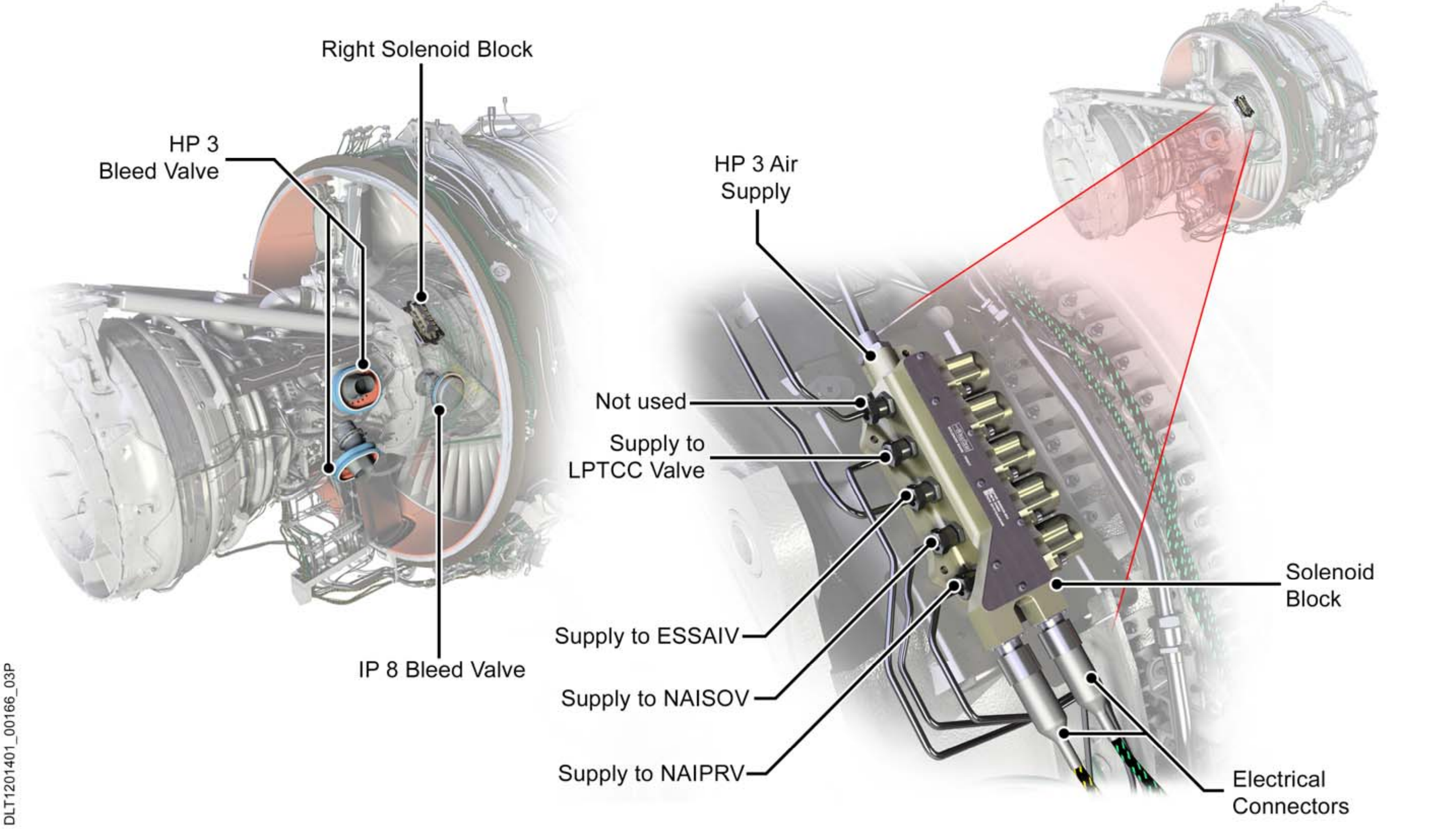


LEFT HAND SOLENOID BLOCK

Right solenoid block

The right solenoid block operates the components that follow:

- Low Pressure Turbine Case Cooling (LPTCC) Valve.
- Nacelle Anti-Ice Pressure Regulating Shut Off Valve. (NAIPRSOV) open / close.
- Nacelle Anti-Ice Pressure Regulating Shut Off Valve. (NAIPRSOV) regulating.
- Engine Section Stator Anti-Ice Valve.(ESSAIV).
- Each of the solenoid valves can be independently operated with signals from the EEC.

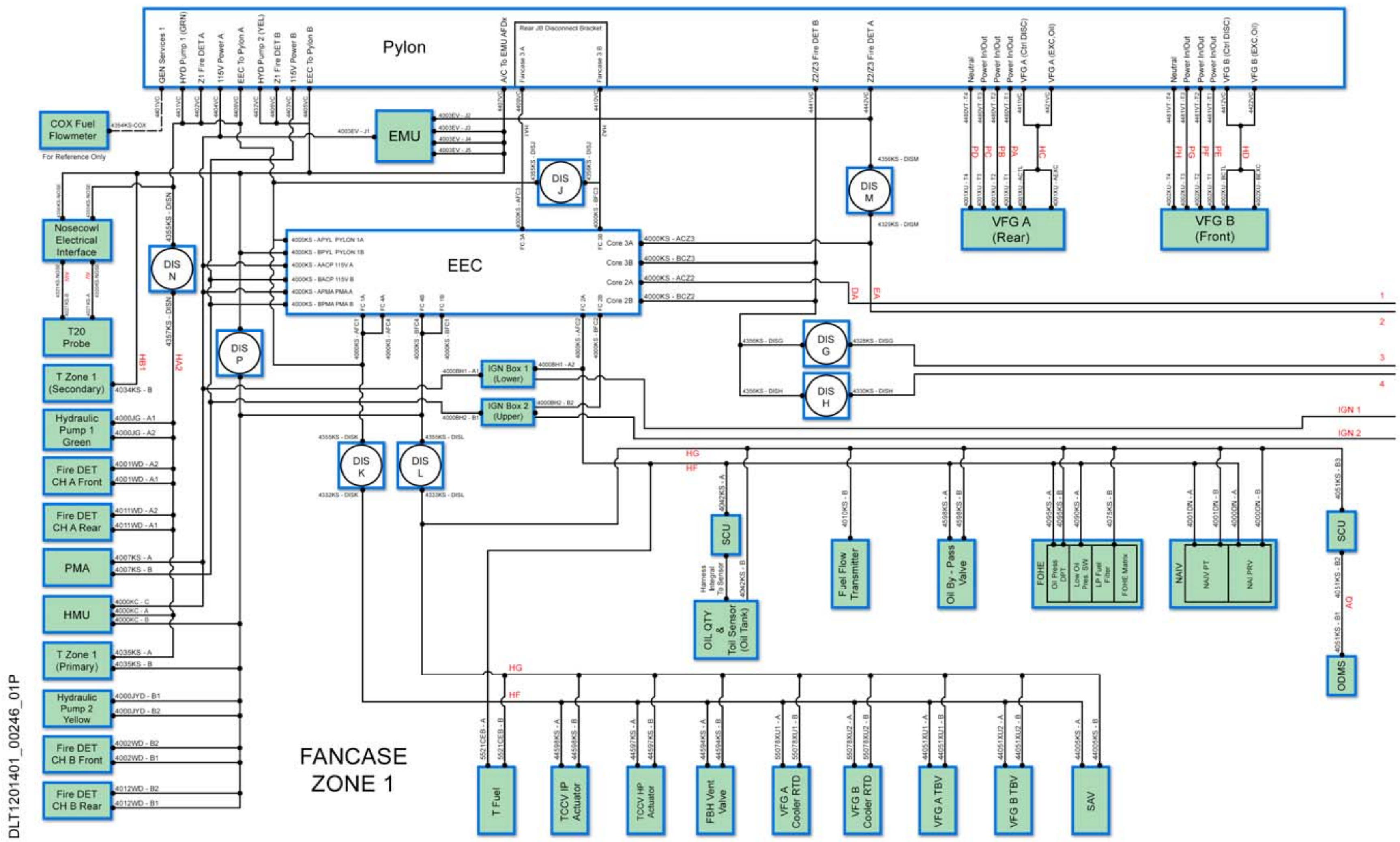


RIGHT HAND SOLENOID BLOCK

Air System Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.

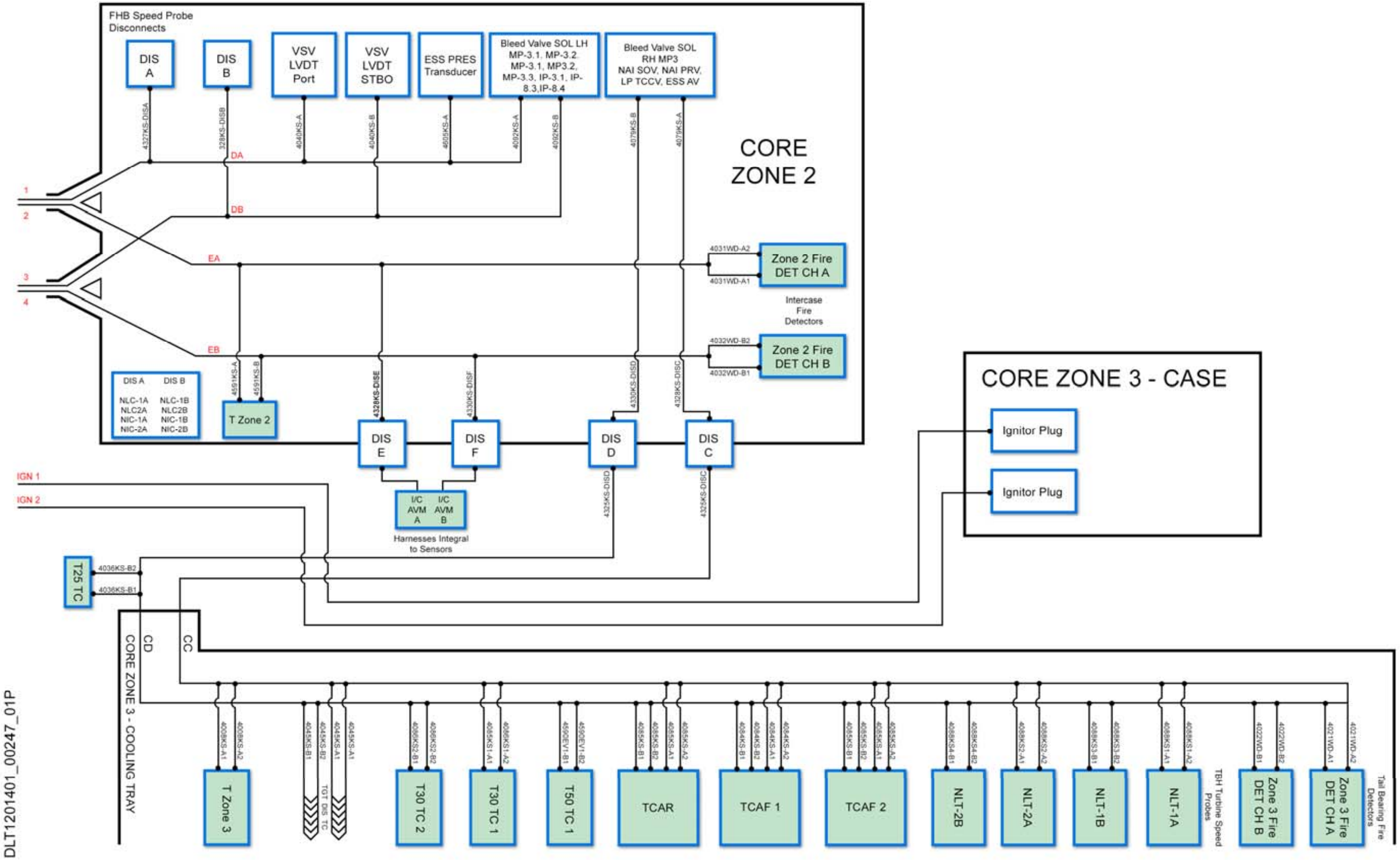


ZONE 1 WIRING DIAGRAM

Air Systems Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.



ZONE 2 & 3 WIRING DIAGRAM

Section 9 - Engine Airflow Control System

Objectives

The student should now be able to:

- State the purpose of the engine Airflow Control System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Airflow Control System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs of the engine Airflow Control System as fitted to the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the engine Airflow Control System as fitted to the Trent XWB engine.
- Describe how the Trent XWB engine Airflow Control System interfaces with other engine and aircraft systems.

End of Engine Air System

Section 10 – Engine Ventilation & Cooling Systems

Section 10 – Engine Ventilation Systems

Aim

At the end of this section the student will be familiar with the Trent XWB ventilation & cooling systems its components, their functions and interfaces.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Ventilation and Cooling System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Ventilation and Cooling System.
- Describe the purpose and operation of the LRUs of the Engine ventilation and Cooling system as fitted to the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Ventilation and Cooling System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Ventilation and Cooling System interfaces with other engine and aircraft systems.

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Ventilation System

Location

The Trent XWB is divided into three primary fire resistant ventilated, temperature monitored areas called zones:

- Zone 1 Fan case compartment.
- Zone 2 Intermediate compressor compartment.
- Zone 3 Core engine compartment.

Purpose

Ventilation is required to remove flammable / hazardous vapours from the zones and keep the zones at an acceptable working temperature. Temperature sensors measure the temperature in zone 1, 2 and 3 of the engine. The sensors measure temperatures that increase from a hot air duct leakage. High temperatures in these regions can result in LRU or nacelle damage. The output from the sensors is sent to a single channel of the EEC through electrical harnesses. If the input signal is higher than the parameters set in the EEC an advisory warning will be sent to the cockpit.

Description

Zone 1

Zone 1 is the annular space between the fan case and the fan cowl doors. The zone contains the EEC, EMU, and external gearbox, oil and fuel units.

This is a cool zone and is ventilated by ambient ram air through an inlet located on the upper section of the nose cowl and is exhausted through a louvered vent in the right side fan cowl door.

Zone 2

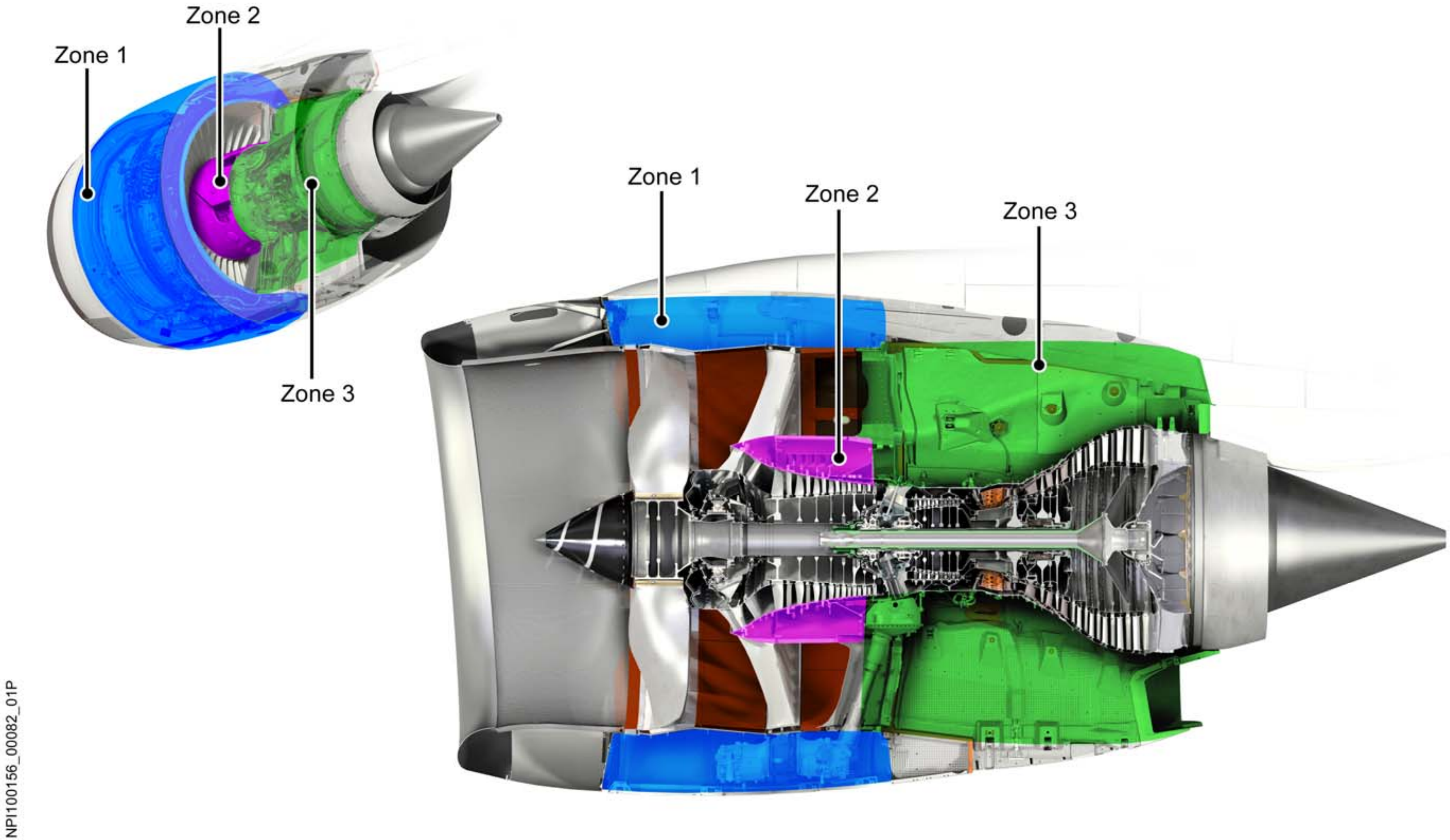
Zone 2 is the annular space between the intermediate (IP) compressor case and below the gas generation core fairings. The zone contains fuel tubes to the VSV actuators and oil tubes.

Fan bypass air enters the zone through a ventilation circular inlet located towards the front of each lower gas generation core fairing. There are vent holes in the lower 30 degrees of the fairings to allow positive ventilation of the Zone 2. Air is then exhausted from the zone through louvers in each upper gas generator fairing back into the fan bypass duct.

Zone 3

Zone 3 is the area under the thrust reverser inner fixed structure, from the zone 2 bulkhead to the rear of the LP turbine case. It is divided into four sub-zones described as zones 3A, 3B, 3C and 3D. The zone contains compressor off take ducts, fuel and oil tubes.

Fan bypass air enters the zone 3 through twelve ventilation inlet holes located around the thrust reverser inner fixed structure. Zone 3 is also supplied with air, which is exhausted from the fan air pre-cooler, and the turbine case cooling ducts. The air is exhausted through the rear of the annular nacelle structure.



ZONES 1, 2 & 3

Zone 1 Temperature Sensors

Location

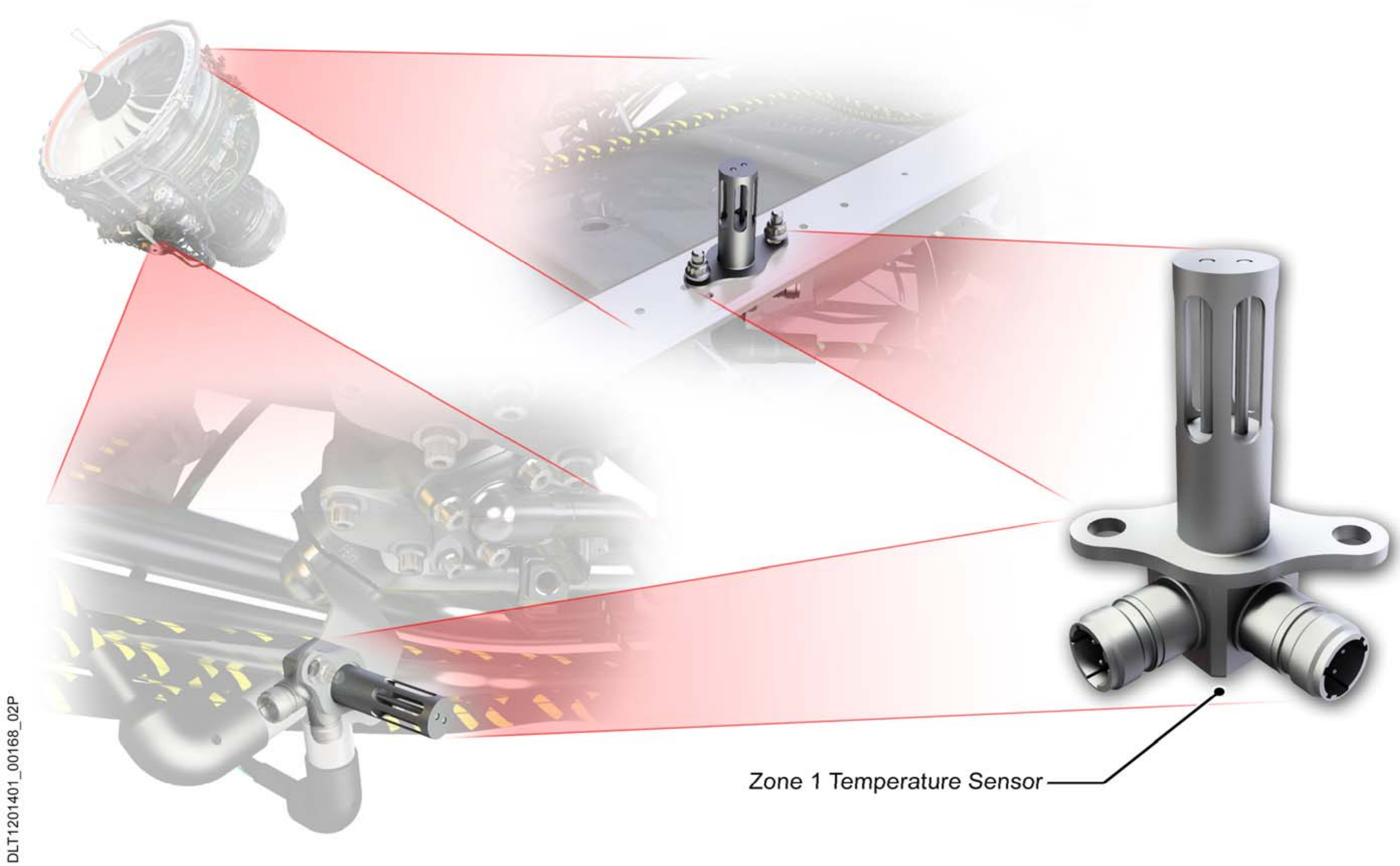
There are two zone 1 temperature sensors one is attached to a bracket on the lower side of the external gearbox mounted oil pump. The other one is mounted on a bracket near the 11 o'clock position of the Fan Case.

Purpose

The purposes of the zone 1 temperature sensors are to detect an anti-ice or starter duct burst event under the nacelle fan cowl doors.

Description

The zone 1 temperature sensors are a resistance temperature device (RTD). The RTD consists of two elements each providing independent signals to the EEC through electrical harnesses. If the input signal to the EEC becomes higher than a preset limit an advisory cockpit indication will be generated.



ZONE 1 LOWER TEMPERATURE SENSOR

Zone 2 Temperature Sensor

Location

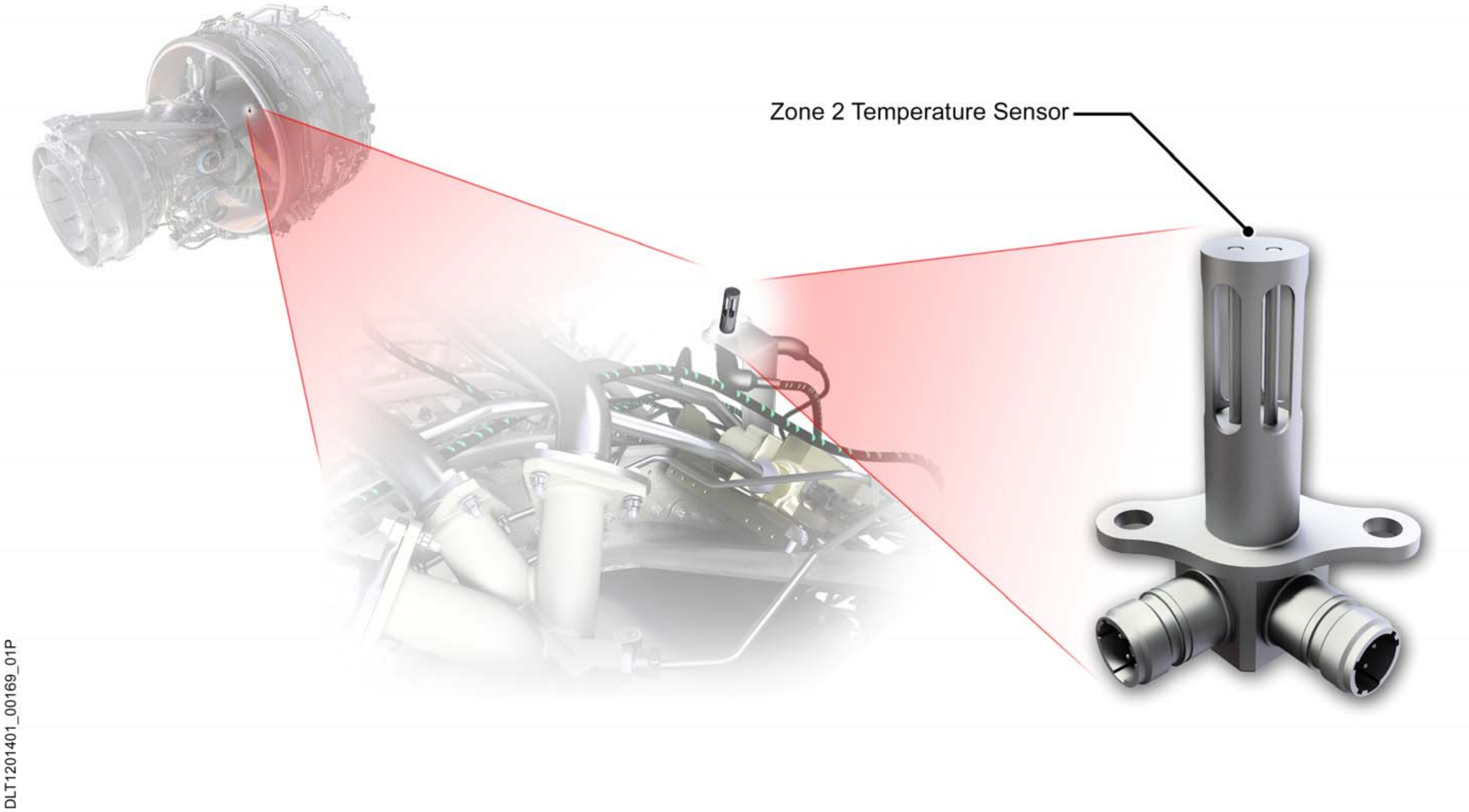
The zone 2 temperature sensor is attached to a bracket on the left side of the engine at the 1 o’clock position in zone 2.

Purpose

The purpose of the zone 2 temperature sensor is to detect an ESS anti-ice duct burst event under the Intermediate compressor gas generator core fairings.

Description

The zone 2 temperature sensor is a resistance temperature device (RTD). The RTD consists of two elements each providing independent signals to the EEC through electrical harnesses (Channel ‘A’ & ‘B’). If the input signal to the EEC becomes higher than a preset limit, an advisory cockpit indication will be generated.



ZONE 2 TEMPERATURE SENSOR

Zone 3 Temperature Sensor

Location

The zone 3 temperature sensor is attached to bracket on the left side of the core engines accessory tray.

Purpose

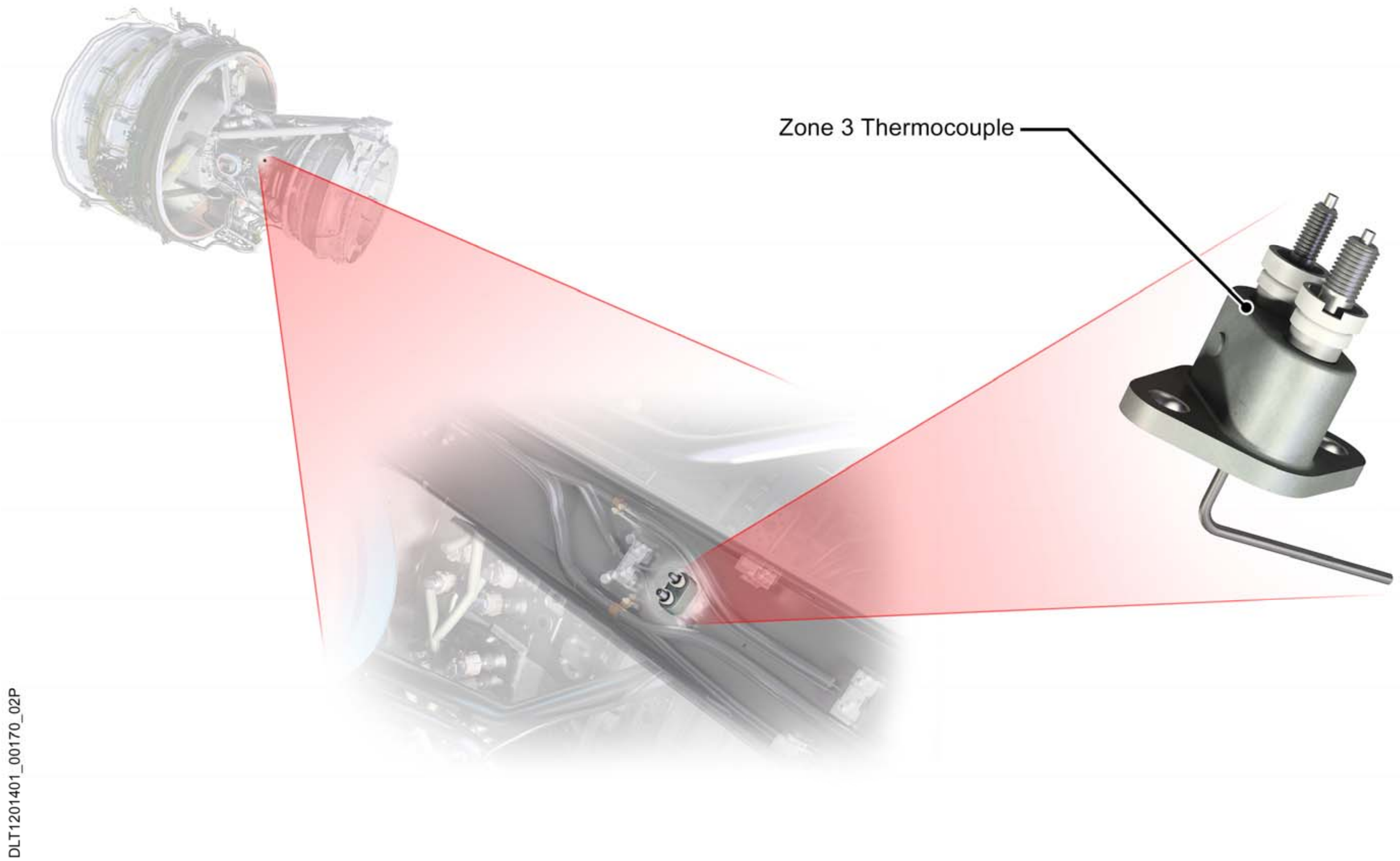
The purpose of the zone 3 temperature sensor is to detect an anti-ice duct burst event under the thrust reverser nacelle structure.

Description

The zone 3 temperature sensor is a thermocouple consisting of two sheathed elements. The output from the thermocouple is sent to a single channel of the EEC through electrical harnesses. An advisory cockpit indication is generated, at a predetermined temperatures limit is attained.

Nacelle Temperature (NAC)

The zone 3 temperature is used as reference for NAC temperature. During normal operation Analogue and digital indications show the temperature of the nacelle temperature in degrees Celsius (C) on the ECAM secondary parameter screen. During engine start this indication will not be seen.



ZONE 3 TEMPERATURE SENSOR

Sealing and Cooling Core

The internal cooling system uses the air from different sources to seal and cool parts of the engine. The compressor air is supplied to different areas of the engine using external rigid tubes and internal openings. The compressor air and the areas where the air is used are as follows

Stage 5 IP Compressor air is used in the areas that follow:

- To cool and pressurise the rear fan seal
- To pressurise and seal the IPC front seal
- To pressurise the front bearing housing.

Stage 8 IP Compressor air is used in the areas that follow:

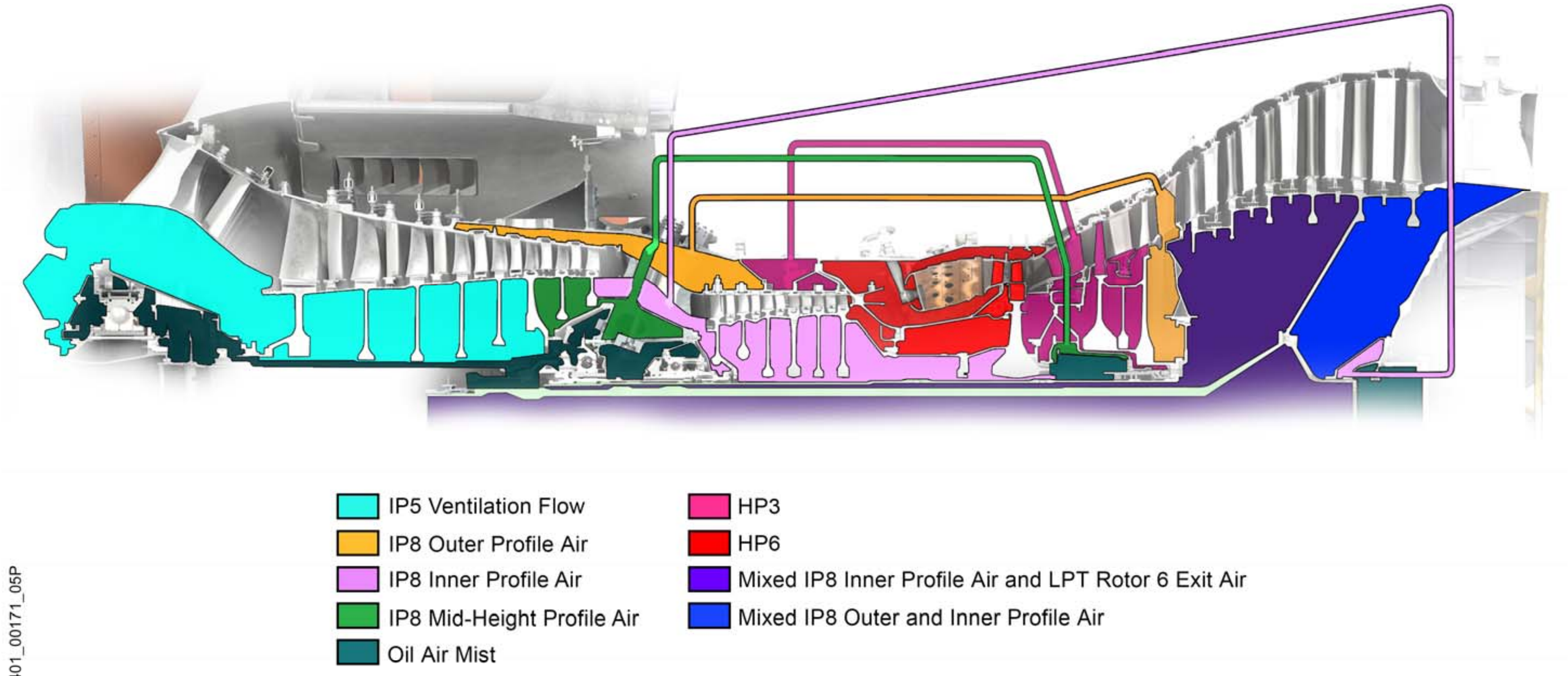
- To cool the HP compressor drum
- To cool the HP turbine disc bore
- To cool the interstages of the of the LP turbine
- To pressurise the IPT2 cavity

Stage 3 HP Compressor air is used in the areas that follow:

- To cool the IP turbine blades
- To cool the IP turbine 1 nozzle guide Vanes (NGVs)
- To seal the HP turbine rear face, IP turbine 1 and 2 front rims and IP turbine rear face.
- To cool the IP turbine seal segments.

Stage 6 HP Compressor air is used in the areas that follow:

- To cool the HP turbine blade
- To ventilate the combustion chamber
- To cool the HP turbine seal segments



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COOLING & SEALING AIR

Turbine Case Cooling (TCC) System

Location

The Turbine Case Cooling (TCC) system is located in various positions on the outside of both the Fan and core case.

Purpose

The Turbine Case Cooling (TCC) system cools the engine HP, IP and LP turbine cases.

Cooling the turbine cases maintains their integrity and controls the turbine blade tip clearances, thus so maintaining turbine efficiency and better fuel consumption.

Description

The TCC system comprises of the following items:

- LP pneumatic operated control valve
- HP, IP mechanical operated control valves
- Distribution manifolds
- HP, IP fuel operated actuators
- HP, IP control cables
- Engine Electronic Controller (EEC)
- HMU (for HP & IP actuator hydraulic fuel medium)
- Solenoid (RH solenoid block)

The C-duct fan / bypass air is the cooling medium used for the 3 turbine cases; this cooling air is controlled by the LP, IP and HP TCC valves to the manifolds.

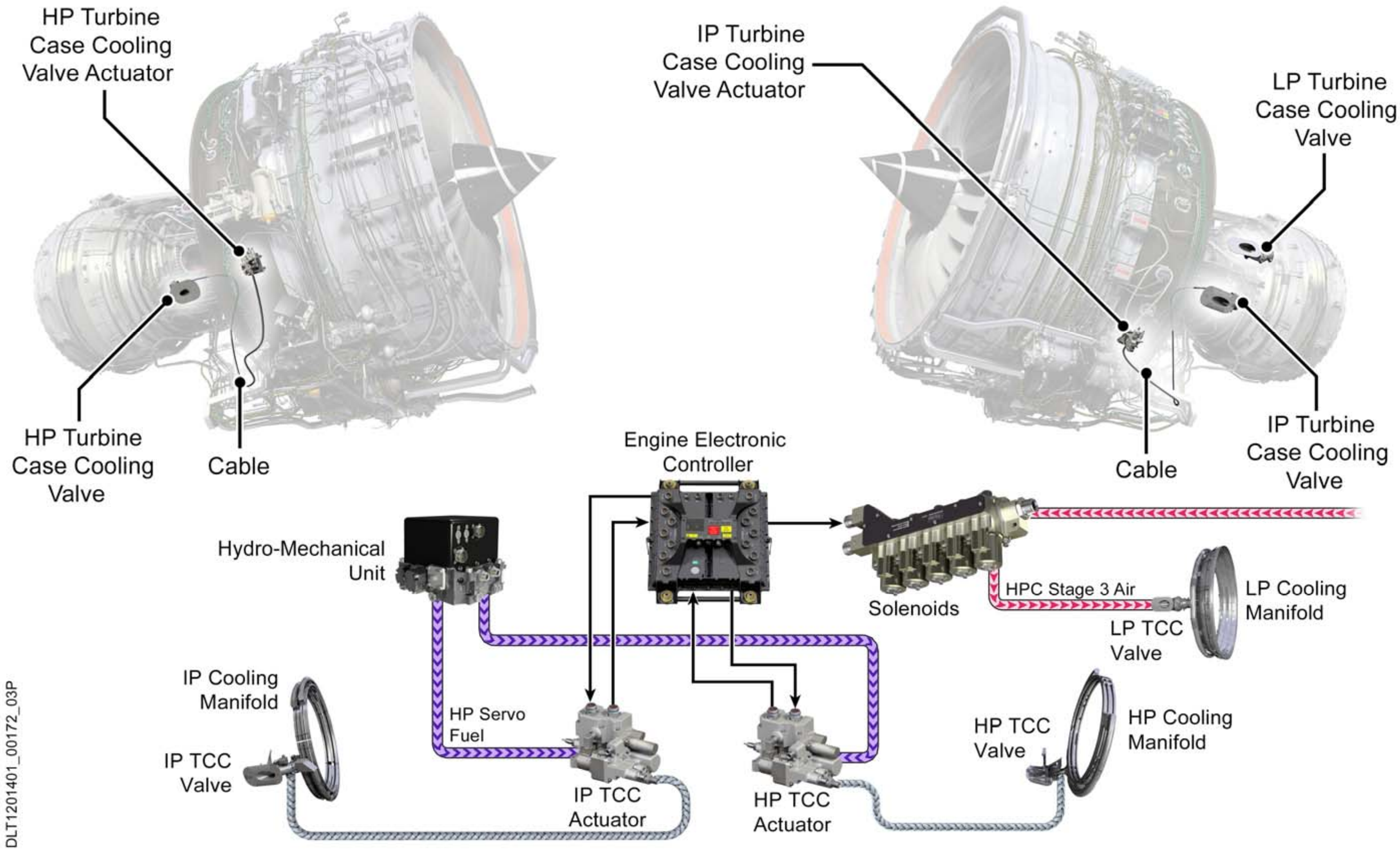
The three manifolds are located externally around their respective LP, IP and HP turbine cases.

The pneumatically operated LP TCC valve is automatically controlled by the EEC through the right hand side solenoid block. In usual operation the valve is open in the cruise condition and closed at all other conditions. Although the valve is in the closed position, a decreased flow of cooling air is delivered to the LP TCC air duct. The EEC uses N1 (LP compressor speed), altitude and the rate of change of altitude to control the LP TCC valve.

The IP and HP control valves are fully modulating and automatically controlled by the EEC for varying flight conditions. The valves are mechanically operated remotely by cables from HMU HP fuel driven actuators.

The temperature of the airflow around the IP stage one and two turbine discs is monitored by two thermocouples. They sense air temperature at front of the IP turbine disc stage one (TCAF), and the temperature at the rear of the IP turbine disc stage two (TCAR). The EEC compares these temperatures against computed IP compressor exit temperature (T25) and altitude to control the IP and HP TCC valves.

If the EEC solenoid fails, the valves will close and as a result there will be a small loss in performance.



TURBINE CASE COOLING SYSTEM

IP/HP Turbine Case Cooling Component Description

IP/HP Turbine Case Cooling Valve

Location

The IP TCC valve is located on the lower left side of the HP turbine 7 o'clock position.

The HP TCC valve is located on the lower right side of the HP turbine 5 o'clock position.

Operation / Control and Indicating

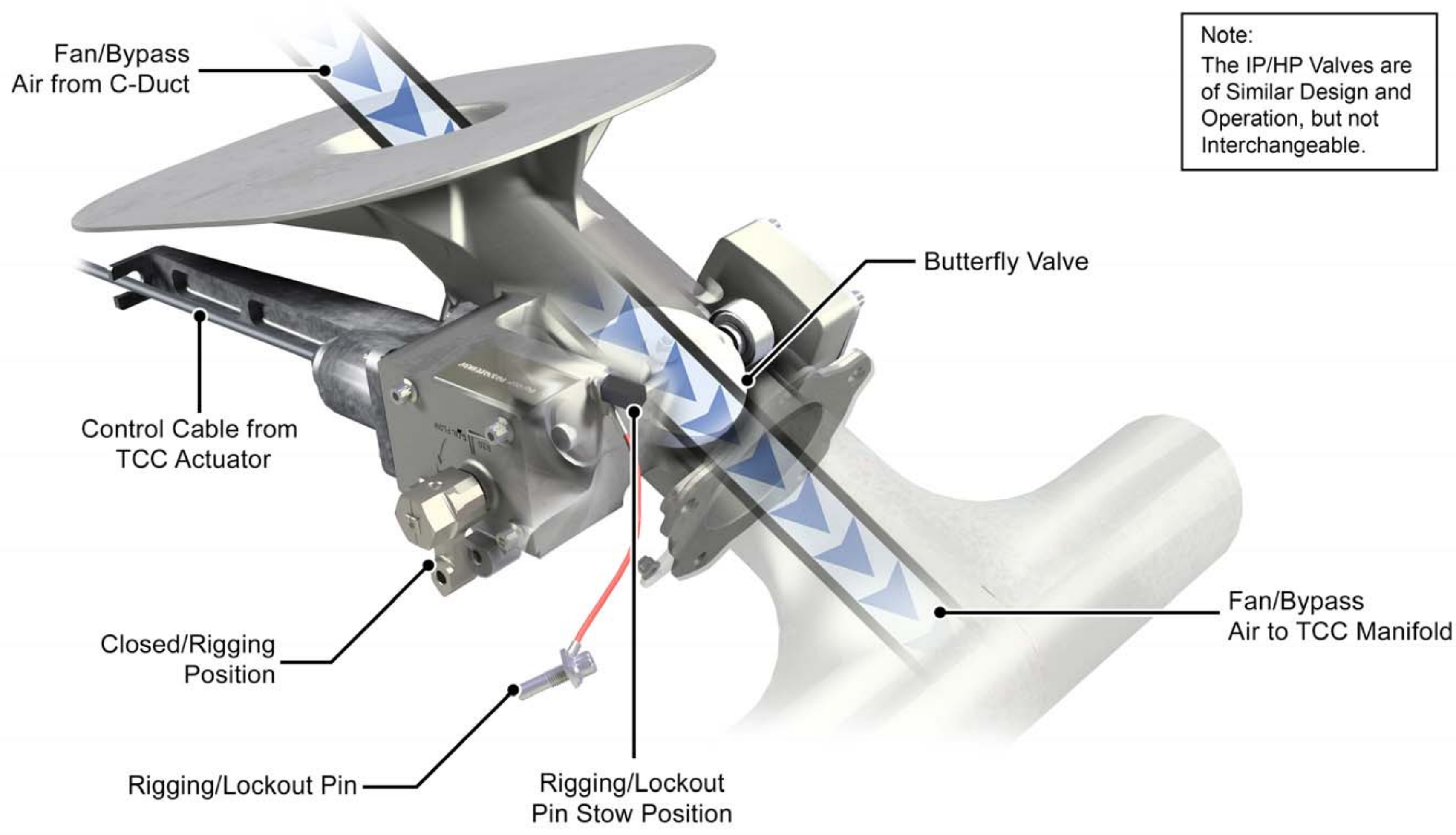
The function of the IP / HP TCC valves is to control a supply fan/bypass air to the IP / HP TCC manifolds.

The butterfly type valve is operated remotely via a control cable attached to the IP / HP TCC controller actuator whose operation is controlled electronically by the EEC.

The actuators use HP fuel supplied by the Hydro Mechanical Unit (HMU) as a force to move a cable that connects to the IP/HP TCC valves.

The linear movement of the IP / HP actuators will move the cable to open or close the IP / HP TCC valves.

For rigging or deactivation of the valve in the closed position, the valve pivot shaft and body incorporates a locking location, where a special locking bolt can be inserted. When not in use the bolt is stowed in the valve body stow location.



Note:
The IP/HP Valves are of Similar Design and Operation, but not Interchangeable.

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IP / HP TCC VALVE

IP / HP Turbine Case Cooling Valve Controller Actuator

Location

The IP TCC valve Controller actuator is located on the lower left side of the fan case 7 o'clock position.

The HP TCC valve Controller actuator is located on the lower right side of the fan case 5 o'clock position.

Operation / Control and Indicating

The function of the IP / HP turbine case cooling valve Controller actuators is to open and close the IP / HP TCC valves via a control cable.

The Controller actuators are operated by fuel pressure supplied by rigid tubes from the Hydro Mechanical Unit (HMU) that is controlled by the EEC. The EEC monitors the position of the actuators in all conditions, which in turn relates the valve positions.

With no fuel pressure, the Controller actuators spring pressure will move the actuators to fully extended position (Fail safe) to close the TCC valves.

Actuators shaft extension feedback to the EEC is produced by the actuators LVDT and is used by the EEC to reference valves positions.

The following faults will be detected by the hysteresis shaft, within the Controller actuators LVDT:

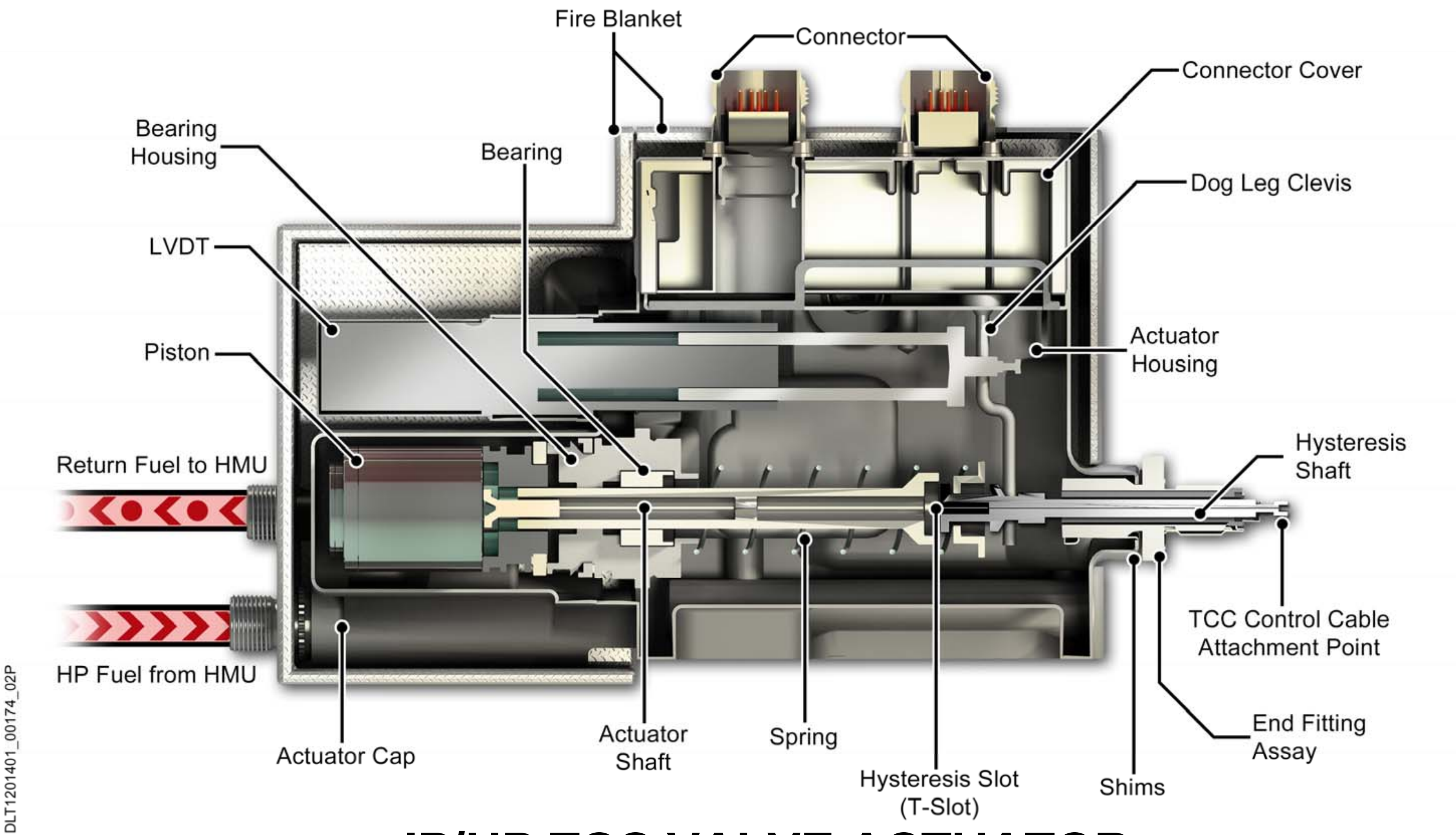
- Cable brake/ disconnection
- Excessive friction.
- Jammed cable or valve
- Missing or failed valve springs.

The LVDTs will flag these anomalies to the EEC as a fault and the EEC will signal the actuator to fully extend in an attempt to close the valve (Fail Safe).

Excess friction in the cable and / or valve, or loss of spring force in the system will result in the loss of tension and so result in a compressive force at the cable / actuator output shaft.

A jammed cable or valve results in a compressive force at the actuator output shaft, which will generate a position error for the system

These conditions will be detectable at least once per flight.



IP/HP TCC VALVE ACTUATOR

LP Turbine Case Cooling Valve

Location

LP Turbine Case Cooling (TCC) valve is located on the left side of the HP & IP Turbine cases at 9 o'clock position.

Operation / Control and Indicating

The function of the LP TCC valve is to supply fan / by-pass air from the C-duct to LP TCC manifold.

The LP TCC valve is pneumatically operated using HPC stage 3 bleed air via a solenoid-operated valve that is part of the right solenoid block, and electronically controlled by the EEC.

When the EEC signals the solenoid valve to energized / open HP3 air will be directed to the LP TCC valve via a rigid tube.

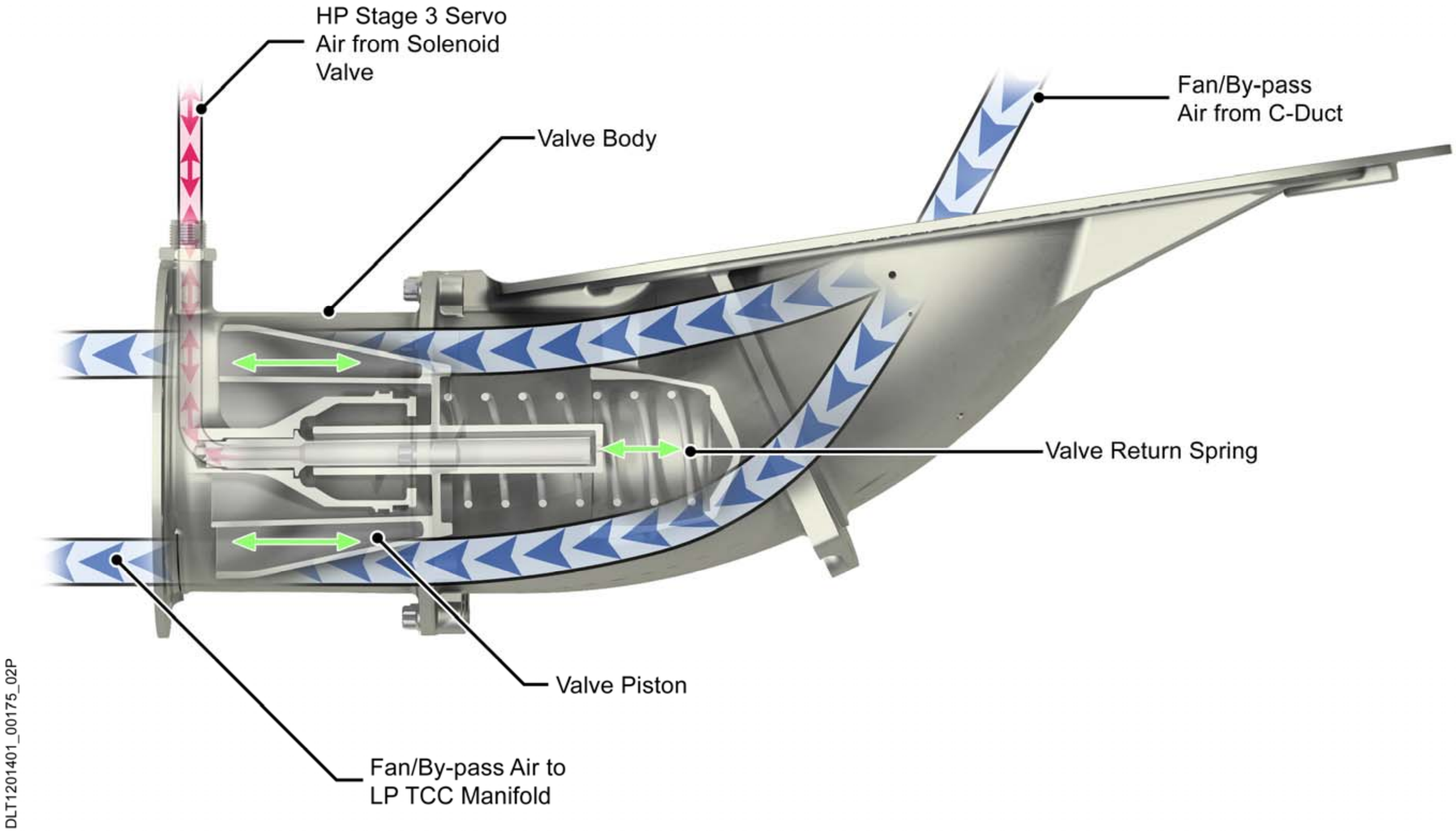
The HP3 air will flow into a piston type valve, push against the spring and move the valve to the fully open position.

When the solenoid valve is de-energized / closed the HP3 air in the solenoid to valve tube, will be released to atmosphere and the valve will move due to spring pressure towards the partially open position / fail safe position.

In usual operation the valve is open in the cruise condition and partially open position at all other conditions.

With the valve in the partially open position, a decreased flow of air is permitted to go into the LP TCC air duct.

The EEC uses N1 (LP compressor speed), altitude and the rate of change of altitude to control the LP TCC valve.


HP Stage 3 Servo Air from Solenoid Valve
Valve Body
Fan/By-pass Air from C-Duct
Valve Return Spring
Valve Piston
Fan/By-pass Air to LP TCC Manifold
DLT1201401_00175_02P

LP TCC VALVE

HP, IP and LPTCC Manifolds

Location

The HP, IP and LP manifolds are located on the core engine around the relative turbine casings.

Purpose

The purpose of the HP, IP and LP cooling manifolds is to circulate fan bypass air around the HP / IP and LP turbine cases and pass it through impingement holes and a duct onto the turbine external cases.

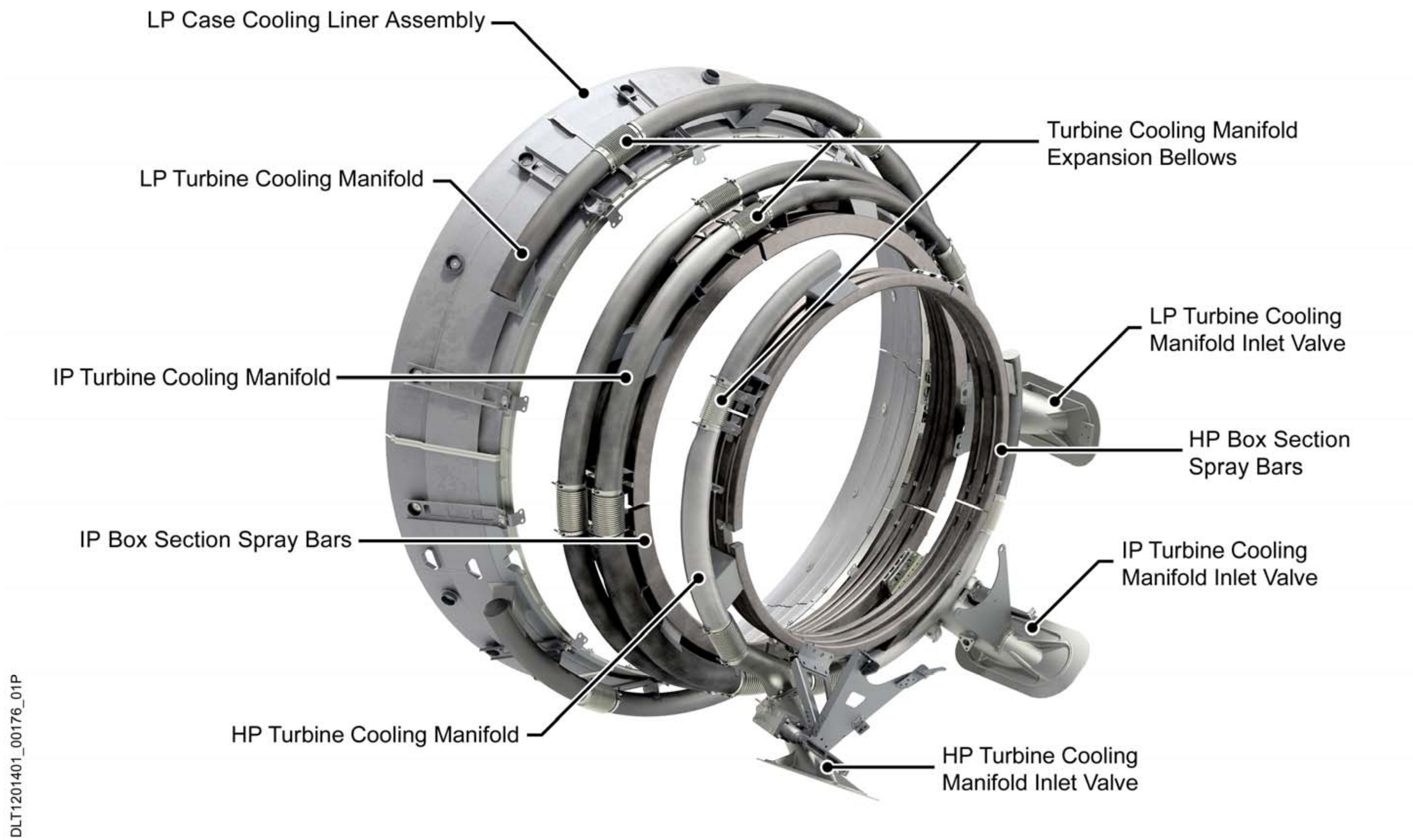
Description

The turbine cases are supplied with Fan / By-pass air from the C-duct through the HP, IP and LP TCC valves to manifolds located around the outside of the HP, IP and LP turbine cases.

The HPTCC manifold comprises of an inlet manifold, a circumferential feed duct with bellows and three box section spray bars with impingement holes.

The IPTCC manifold is comprised of an inlet manifold, two circumferential feed ducts with bellows and four box section spray bars with impingement holes.

The LPTCC manifold is comprised of an inlet manifold, one circumferential feed duct and a case cooling liner assembly.

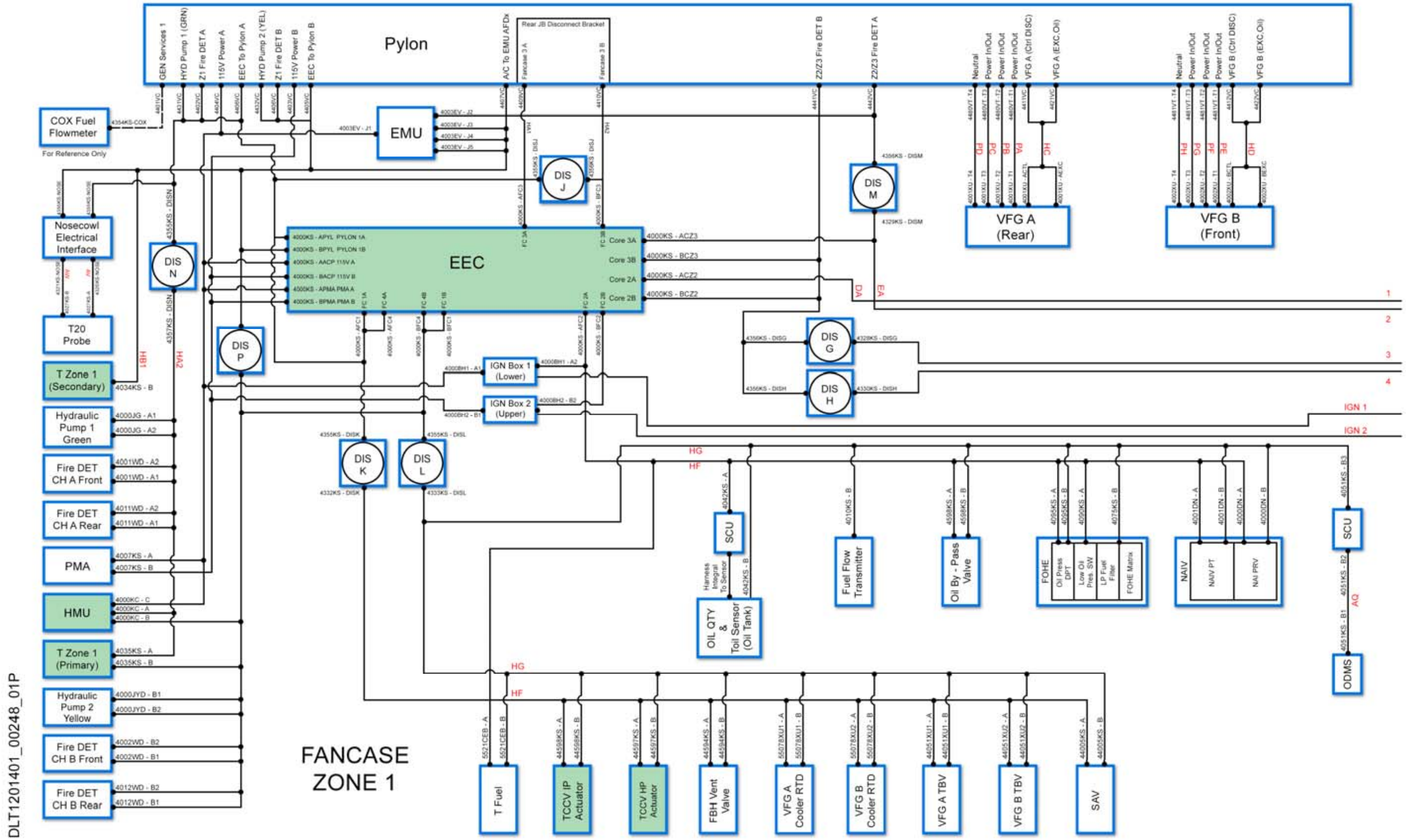


TCC MANIFOLDS

Ventilation and Cooling Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.

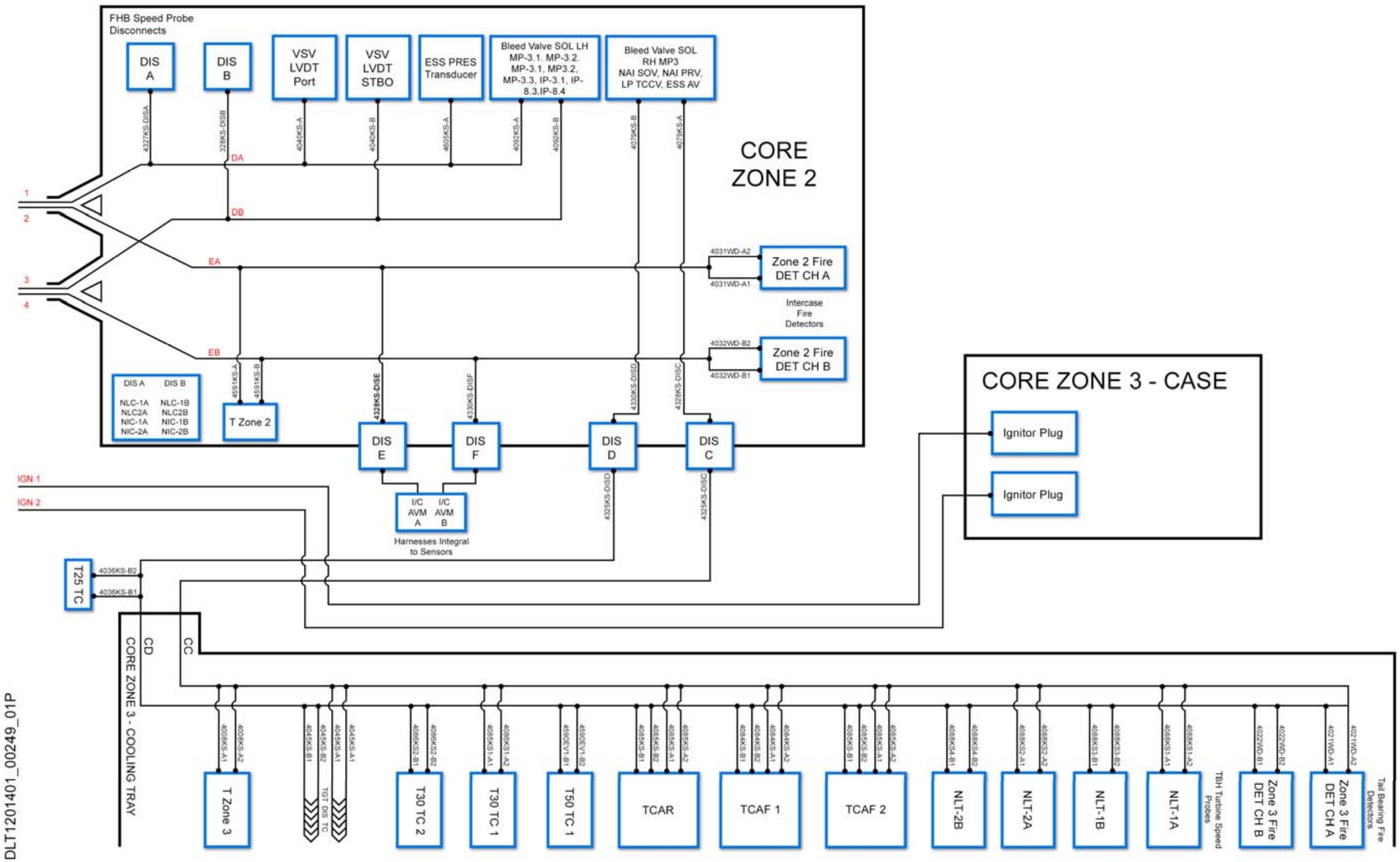


ZONE 1 WIRING DIAGRAM

Ventilation and Cooling Reference Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the official aircraft maintenance documentation.



ZONES 2 & 3 WIRING DIAGRAM

Section 10 – Engine Ventilation Systems

Objectives

The student should now be able to:

- State the purpose of the Engine Ventilation and Cooling System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Ventilation and Cooling System.
- Describe the purpose and operation of the LRUs of the Engine ventilation and Cooling system as fitted to the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Ventilation and Cooling System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Ventilation and Cooling System interfaces with other engine and aircraft systems.

End of Engine Ventilation & Cooling Systems

Section 11 – Engine Fire Protection Systems

Section 11 – Engine Fire Protection Systems

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Fire Protection System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Fire Protection System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Engine Fire Protection System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Fire Protection System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Fire Protection System interfaces with other engine and aircraft systems.

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Engine Fire Protection System

Introduction

Each fire detector monitors a fire zone and senses a possible fire / over heat condition. They send analogue signals to a Conversion Module(CM) (one for each engine), which changes the analogue values into digital values, which are then sent to the related Fire Protection Function (FPF) application hosted in four Core Processing Input / Output Modules (CPIOMs). There are 10 fire detectors each consisting of fire detector sensing element and a responder; the detector is connected to either channel A or B. There two detectors mounted in each of the following locations.

- Zone 1 – Forward side of external gearbox
- Zone 1 – Rear side of external gearbox
- Zone 2 – Lower section of the zone
- Zone 3 – Bottom of pylon above HP system
- Zone 3 – Underside of LP turbine case

Fire / Fault Detection

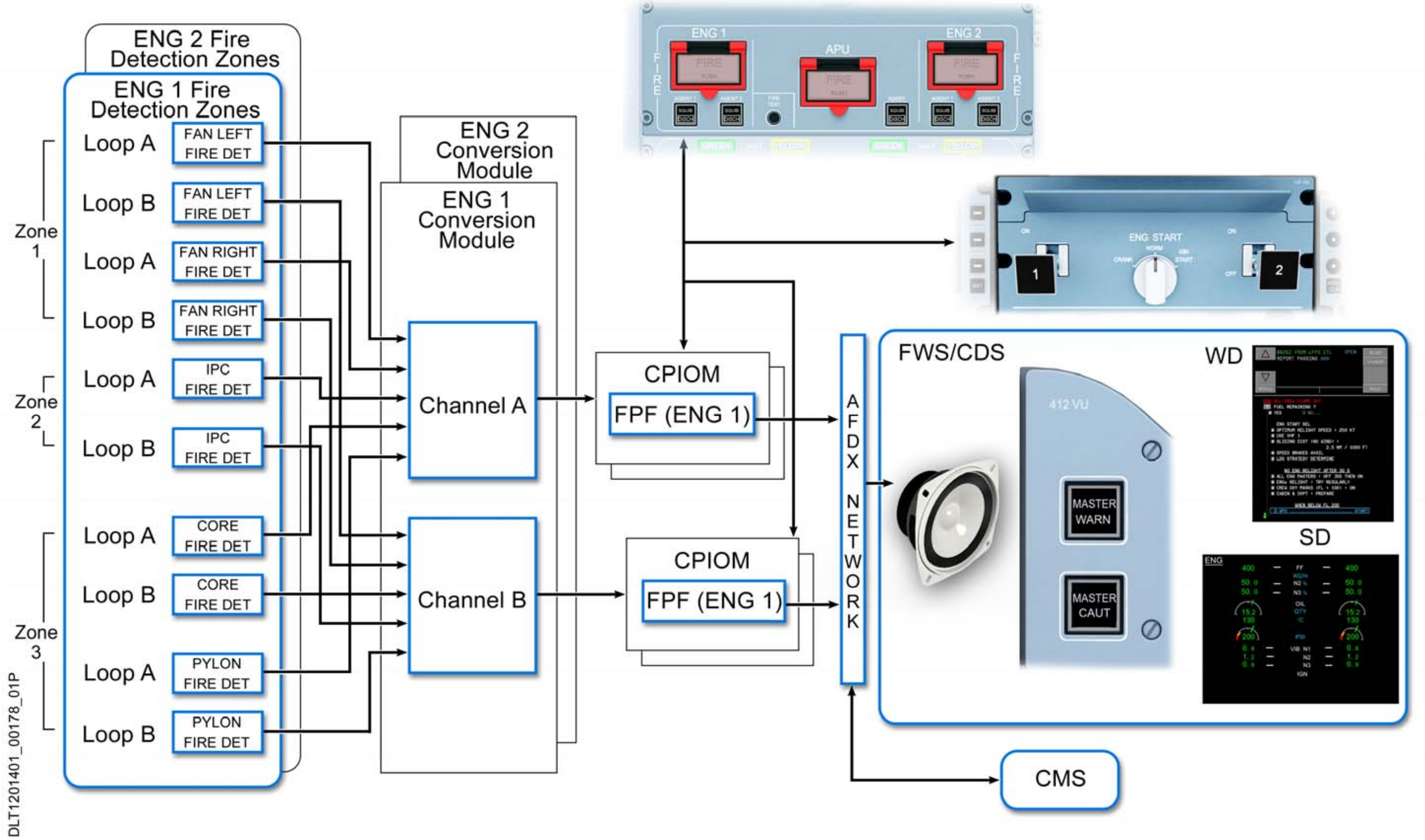
The FPF application hosted in four CPIOMs located in the forward fuselage compartment, collects the digital values from the CM and processes them. If a fire or a fault is sensed in an engine fire zone, the FPF application in the CPIOMs sends a warning to the crew through:

- The Flight Warning System (FWS)
- The FIRE Integrated Control Panel (ICP)
- The ENG MASTER control on the centre pedestal

System Test

The FPF applications are interfaced with the Central Maintenance System (CMS) to do a check of the resistance and continuity of the detectors through a CMS test.

The FIRE TEST P/BSW installed on the FIRE Integrated Control Panel (ICP) is used to do an engine fire operational test in flight or on ground.



ENGINE FIRE DETECTION SYSTEM

Fire / Overheat Detectors

Location

Each detector assembly has an element which is attach to a support tube. Two elements run parallel to each other along with the detectors at opposite ends of the support tube, one detector being channel A and the other being channel B. The elements monitor the temperature along their whole length, providing a continual analogue output to the conversion module for the engine. Quick release clamps and bushing support the elements along their length.

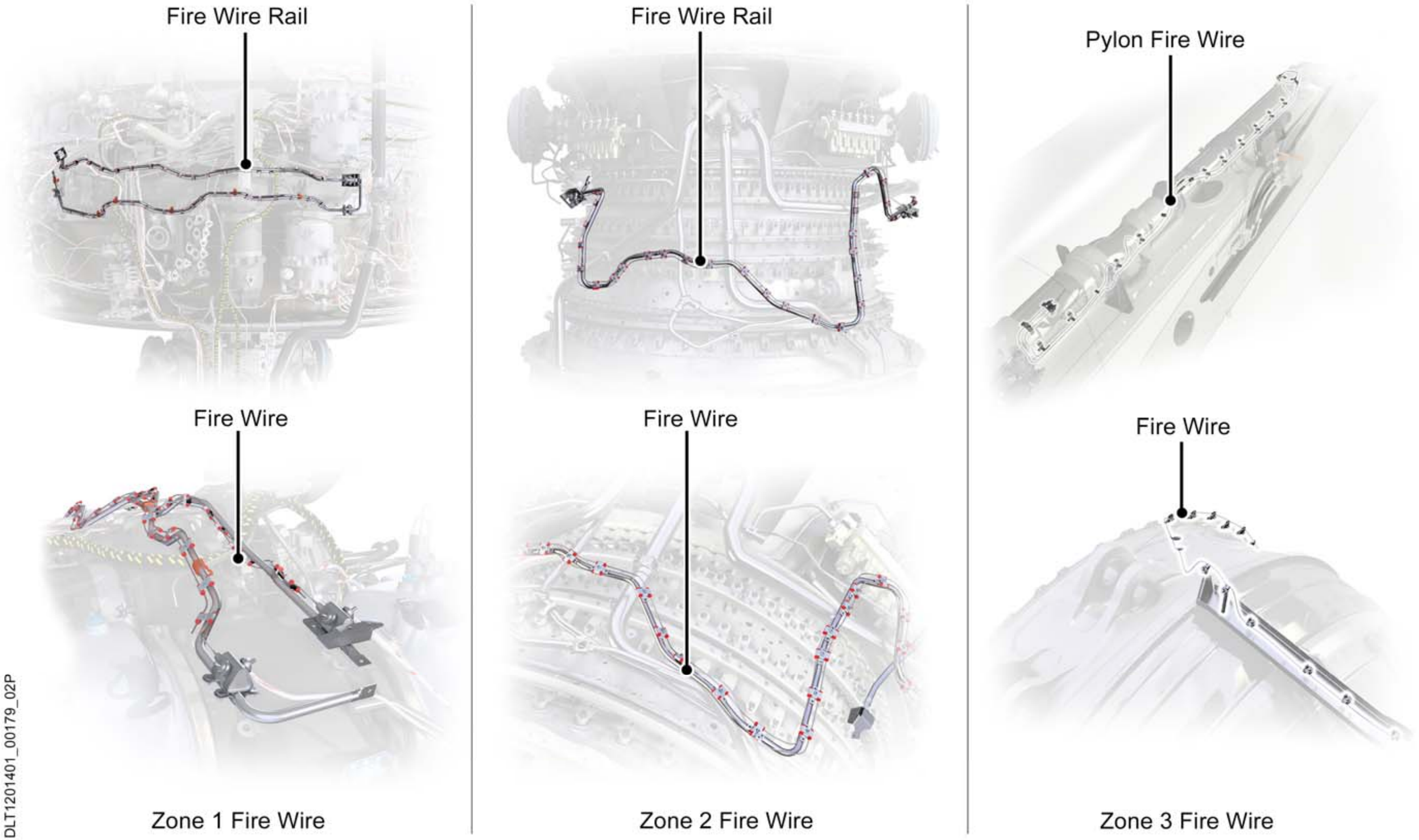
Purpose

The engine fire detection assemblies monitor the temperature in the engine zones.

Description

A fire alarm is triggered if there is;

- A fire detected by the two elements in a common zone.
- One defective element in a zone and the serviceable element detect fire in the same zone.



ENGINE FIRE DETECTION LOOPS

Fire / Overheat Detector Element

Description

The fire detector uses electro-pneumatic technology and consists of a sensing element and a responder assembly.

Sensing Element

The sensing element is a tube, which contains hydrogen charged core. There is a helium filled gap between the core and outer tube. One end of the sensing element is hermetically sealed and the other end is connected to the responder assembly.

Responder Assembly

The responder assembly consists of a stainless steel body. It contains a chamber connected to two pressure switches: an ALARM switch and an INTEGRITY switch. The responder is connected to the aircraft electrical harness. According to the position of the switches, the responder generates three signals that give the state of the fire detectors.

- Normal
- Fault
- Fire

Detector Failure

The integrity switch in the responder unit is kept closed by helium gas pressure. If damage to the sensor causes a gas leak a fault signal is sent to the FPF through the conversion module.

Overheat Detection

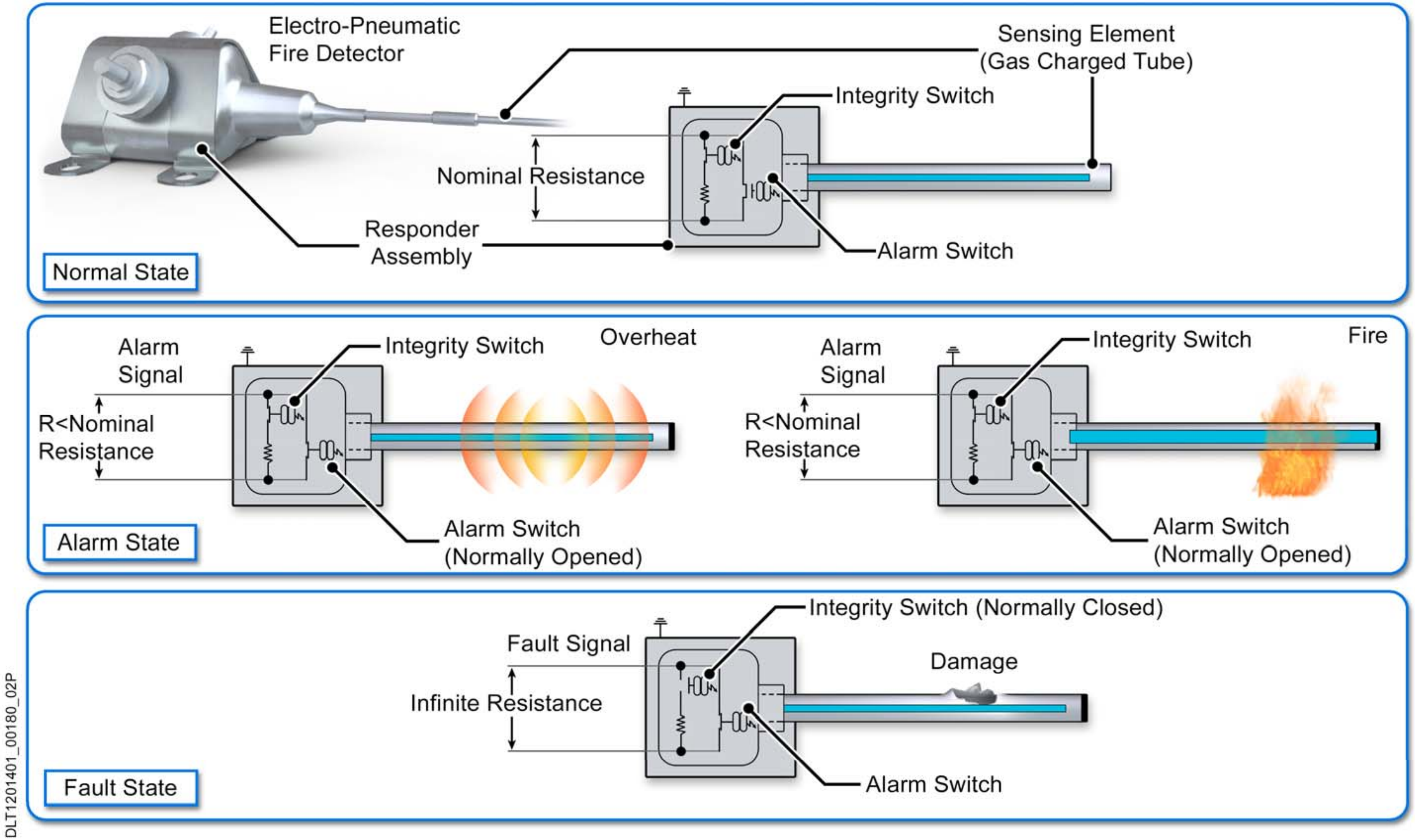
If the air temperature around the detector element increases, the pressure of the gas increases. At a pre-set temperature for the zone, the gas pressure will move the diaphragm to close the alarm switch. A signal is sent to the FPF through the conversion module. The FPF processes the signal and transmits a warning to the cockpit. When the temperature decreases the system returns to normal.

Fire Detection

In a fire condition a smaller section of the detector element will get a large increase in temperature. When it is greater than a pre-set temperature, hydrogen gas is released from the core element. This increases the pressure in the sensor tube and moves the diaphragm to close the alarm switch and a signal is sent to the FPF through the conversion module. The FPF processes the signal and transmits a warning to the cockpit. When the temperature decreases the system returns to normal.

Note:

The responder unit terminal posts and associated cable ends are of a different size to prevent incorrect fitting.



ENGINE FIRE DETECTION SYSTEM

Engine fire-extinguishing system

Purpose

The system has two main functions:

- To extinguish any fire occurring in the engine nacelles.
- To prevent an engine fire from spreading. The engine is isolated from other systems e.g. air, fuel, hydraulics and electrical power.

Description

Each engine fire-extinguishing system has the following components in the system:

- Two fire extinguisher bottles (Halon Gas)
- One conversion module
- FPF within the CPIOMs
- The ENG / FIRE Integrated Control Panel (ICP)
- The TEST pushbutton switch on the ENG / FIRE ICP
- The ENG MASTER Control panel

Conversion Module / FPF

The conversion module receives the input from the pressure switch, which it digitizes and then sends on to the FPF. The FPF uses this input and sends a discharge signal to the DISCH legend on the AGENT pushbutton switch on the ICP.

ENG / FIRE ICP

On the ENG / FIRE ICP there are 2 ENG / FIRE pushbutton switches (one for each engine) and 4 AGENT pushbutton switches (two for each engine)

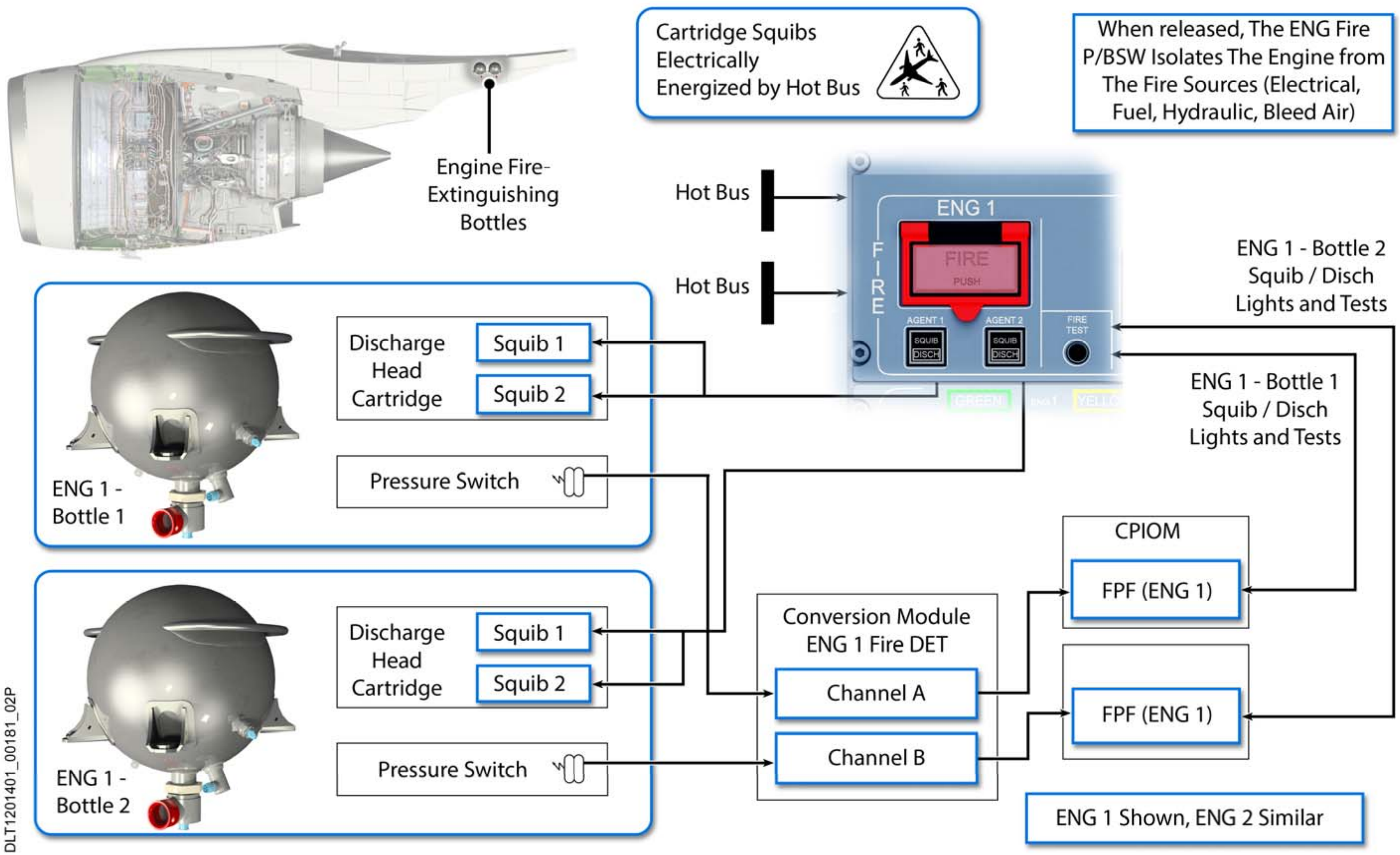
The ENG / FIRE pushbutton switches have 4 main functions:

- It indicates the fire warning generated by the FPF
- It activates the micro-switches involved in the extinguishing procedure
- It arms the discharge function for the fire bottles
- To isolate the engine from the possible fire sources (electrical, fuel, hydraulic, bleed air).

There are two AGENT pushbutton switches for each engine, one for each fire bottle. They indicate when the bottles are armed (SQUIB light up) and when the bottles are discharged (DISCH in amber). When pushed they operate the cartridge in the bottle discharge head.

TEST Pushbutton Switch

The switch is used to test the system serviceability.



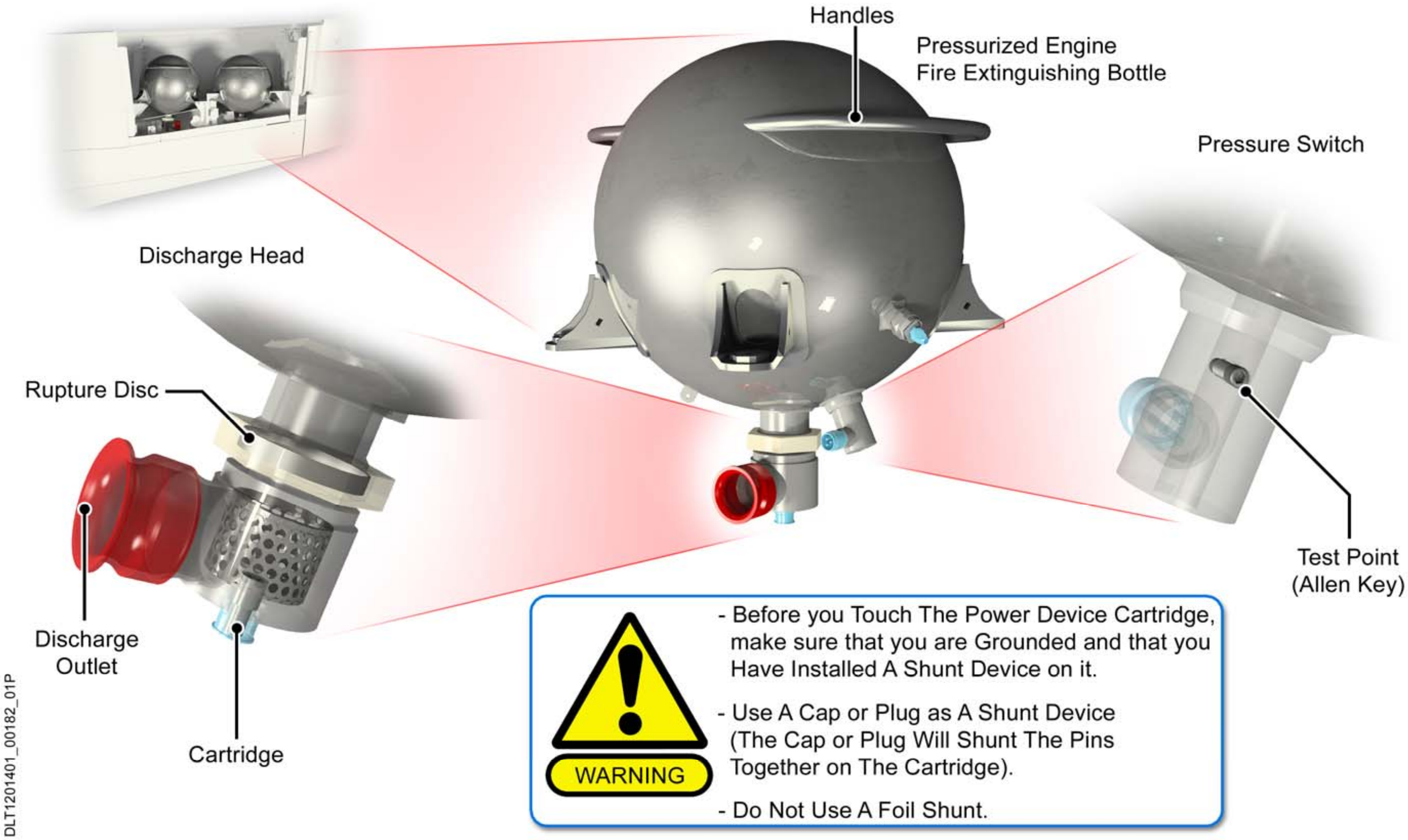
ENGINE FIRE EXTINGUISHING SYSTEM

Engine fire-extinguishing system continued

For each engine there are two fire extinguisher bottles located in the aft section of the pylon. The fire bottles contain pressurized extinguishing agent. Each fire bottle has the following features:

- Two handles to help the removal or installation
- An outlet rupture disc connected to a discharge head outlet
- A cartridge
- A pressure switch for the monitoring.

The integrity of the pressure switch and the low-pressure signal transmission is manually tested. An ALLEN-key test point is used to do this test.



ENGINE FIRE BOTTLES

Engine Fire Detection Conversion Module

Location

The conversion module is located in the mid part of the engine pylon.

Purpose

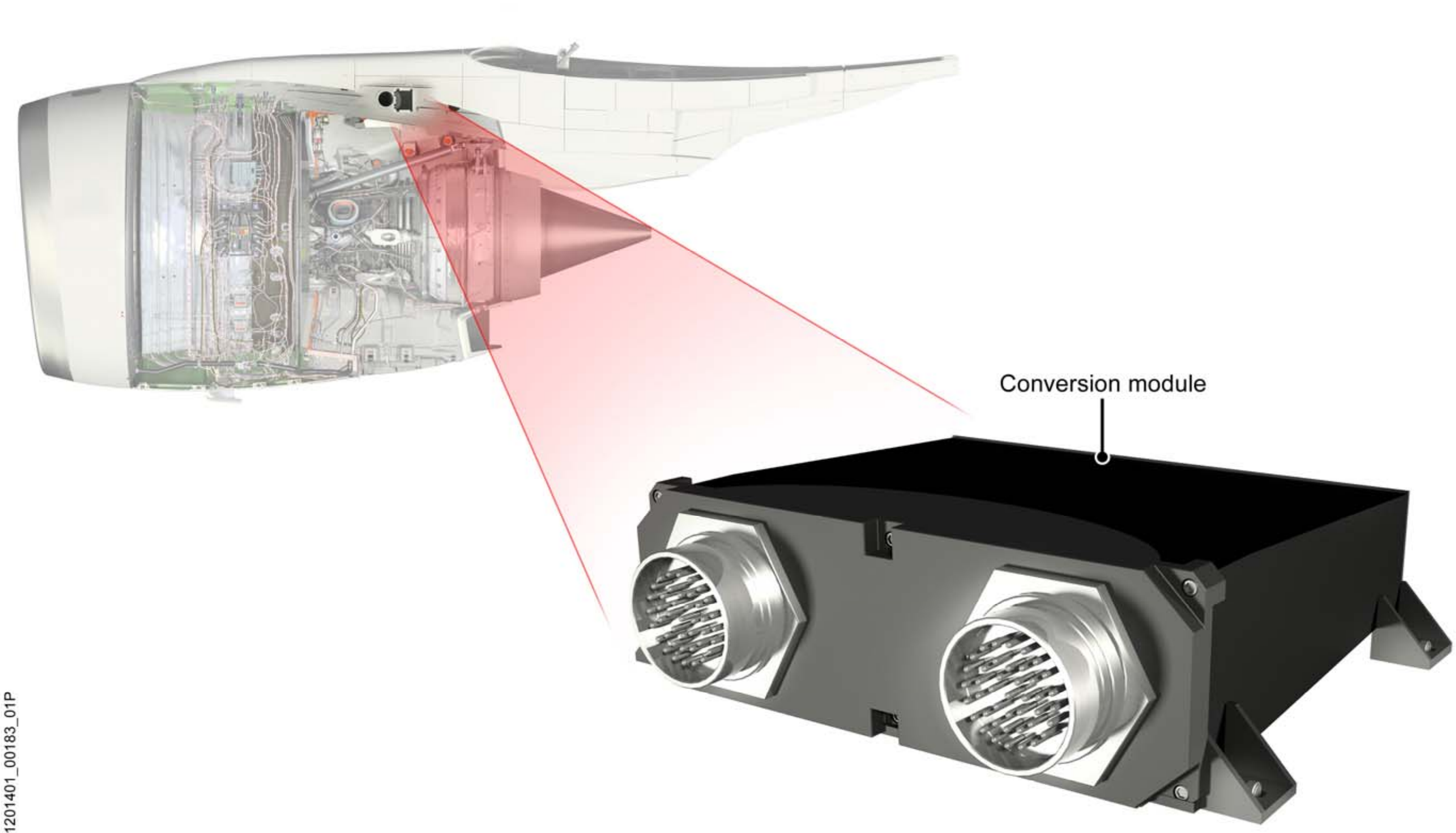
The conversion module digitizes the analogue signals from the fire responders on the engine and outputs these to the FPF within the CPIOMs.

The conversion module also monitors the pressure switch on the fire bottles.

Description

The conversion module has two independent channels A & B, one for each fire loop. Each channel has a separate 28 VDC power supply.

Each channel of the conversion module receives signals from each fire detector responder in its own loop (A or B). These signals are processed and converted to digital signals, which are then sent to the FPF. Each channel also receives signals from the pressure switch of the fire extinguisher bottle. Channel A monitors fire bottle No.1 and Channel B monitors fire bottle No. 2.



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ENGINE FIRE DETECTION CONVERSION MODULE

Fire Zone Integrity

Introduction

The nacelle is designed for fire integrity by the use of firewalls, fire seals, and fire barriers that are incorporated as necessary to isolate fire zones from each other and to isolate fire in the designated fire zone areas and from aircraft primary structure. The engine core compartment is a fire zone bounded by nacelle components and the fire seals.

Pylon

The firewalls in the core compartment are the inner fixed structure, the latch beam, pressure relief doors, the forward fixed panel and the disconnect panel firewall.

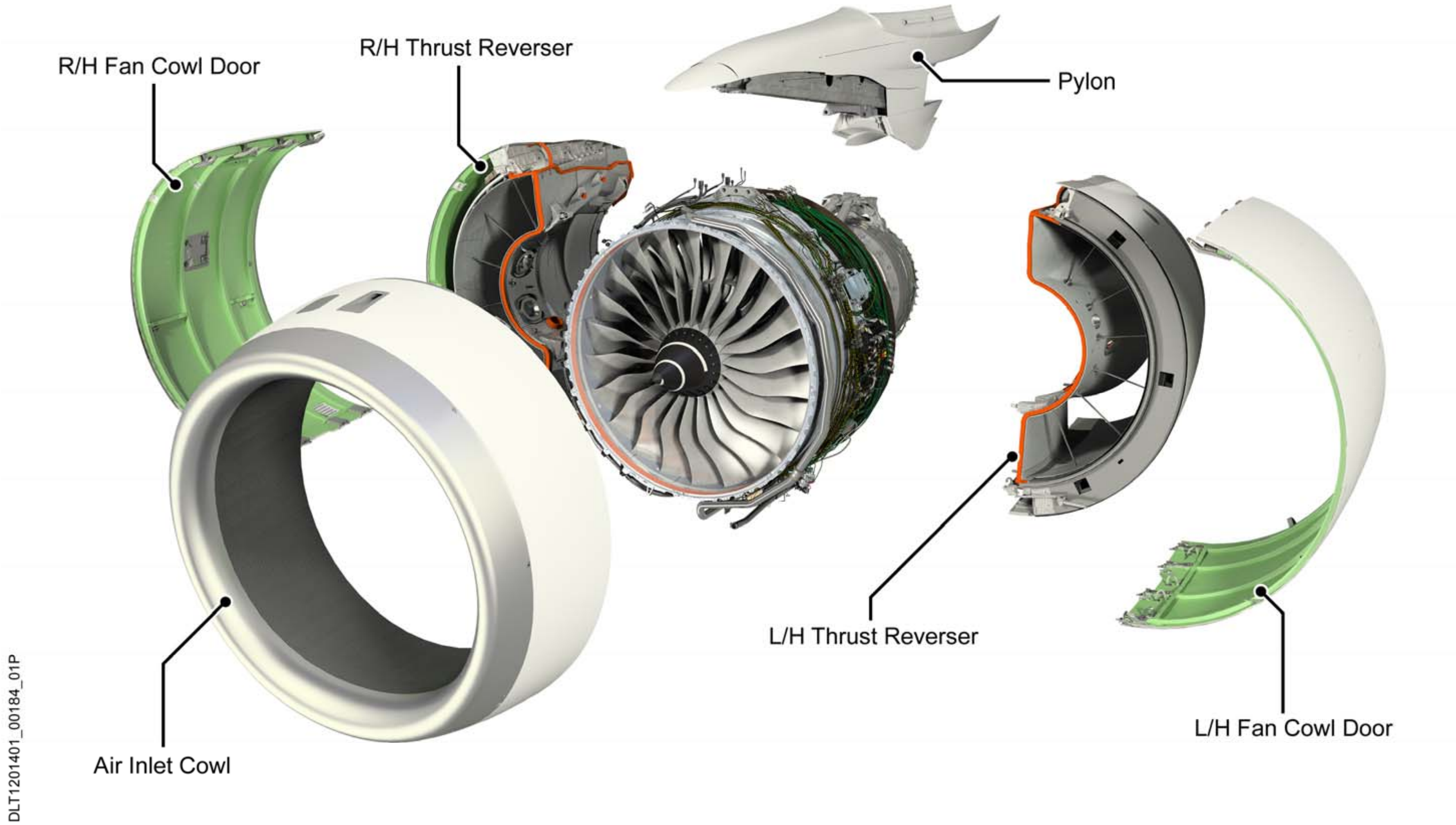
In addition to the firewalls, seals and fire barriers the use of thermal blankets are also used for additional fire and thermal protection.

The Fan Cowls

The fan cowls are the outer boundary of the fan compartment. The fan cowl is a composite structure manufactured from Nextel, titanium and steel, this composite structure gives the fan cowl door a fireproof property.

The Fan Exhaust / Thrust Reverser Cowl

The inner fixed structure panel of the fan exhaust / thrust reverser cowl is designed to be fireproof. This protection is reinforced by the installation of thermal blankets. The inner fixed structure panels of the fan exhaust / thrust reverser cowl are the outer boundary of the core compartment (Zone 3).



FIREPROOF BULKHEADS PANELS AND SEALS

Fire Zone Integrity continued.

High Pressure (HP) 3 Bleed valves

There are three HP 3 bleed valves. Each bleed valve has a fire seal. Each fire seal makes a seal around the thrust reverser inner fixed structure.

Intermediate Pressure (IP) 8 Bleed valves

There are three IP 8 bleed valves. Each bleed valve has a fire seal. Each fire seal makes a seal around the Gas Generation Fairing

Upper Inter-services Fairing

The upper splitter fairing has a horseshoe section seal installed on the lower edge. The seal makes a fireproof seal around the LP compressor case.

Lower Inter-services Fairing

The lower splitter fairing is fireproof. If there is a fire, it helps contain the fire in Zone 3. The fairing has a horseshoe section seal installed on the lower edge. The seal makes a fireproof seal around the LP compressor case.

Bifurcation Panel

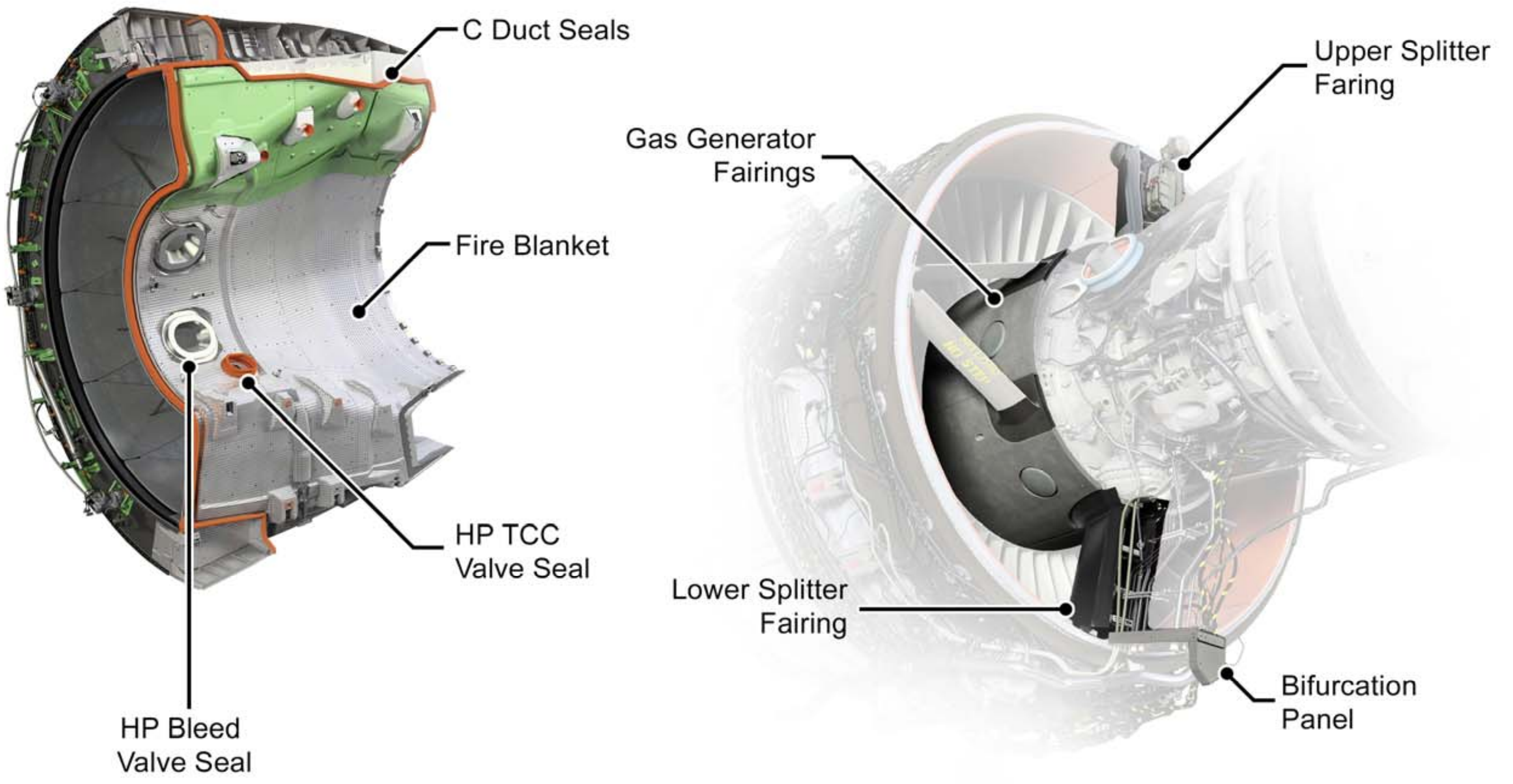
The bifurcation panel has a seal between the panel and the ignition leads. There are also seals between tubes and other various harnesses and the panel.

Compressor Gas Generator Fairings (Core Fairings)

Each gas generator fairing has a D section seal. They are installed on the edges of each gas generator fairing. The D section seal forms a seal against:

- Adjacent gas generator fairings.
- The lower interservices fairing.
- The upper interservices fairing.
- The zone 2 / 3 diaphragm.

These are also sealed with a silicon based sealant (See the relevant maintenance documentation for details).



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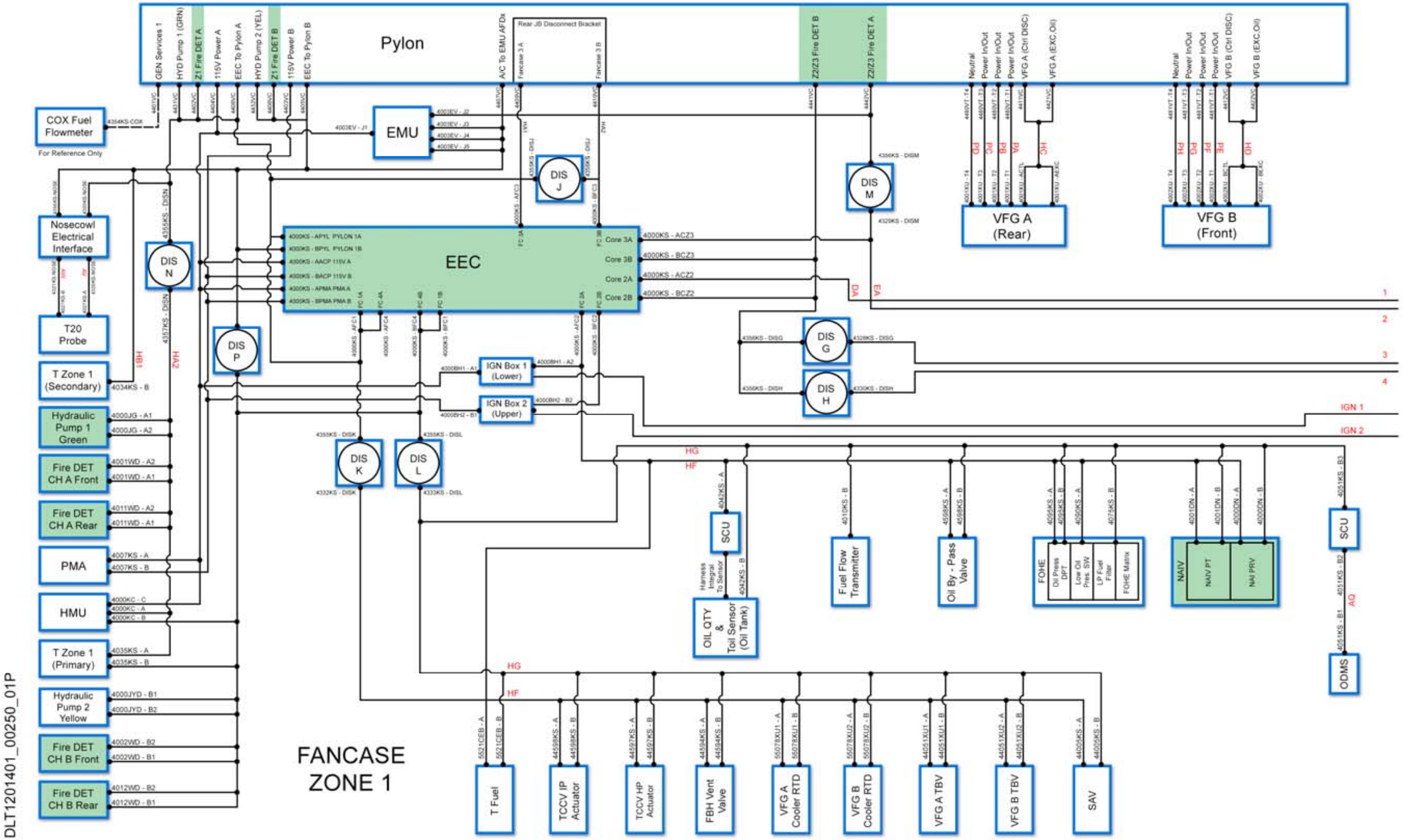
FIREPROOF BULKHEADS PANELS AND SEALS

Ventilation and Fire Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.



ZONE 1 WIRING DIAGRAM

Ventilation and Fire Wiring Diagrams

Introduction

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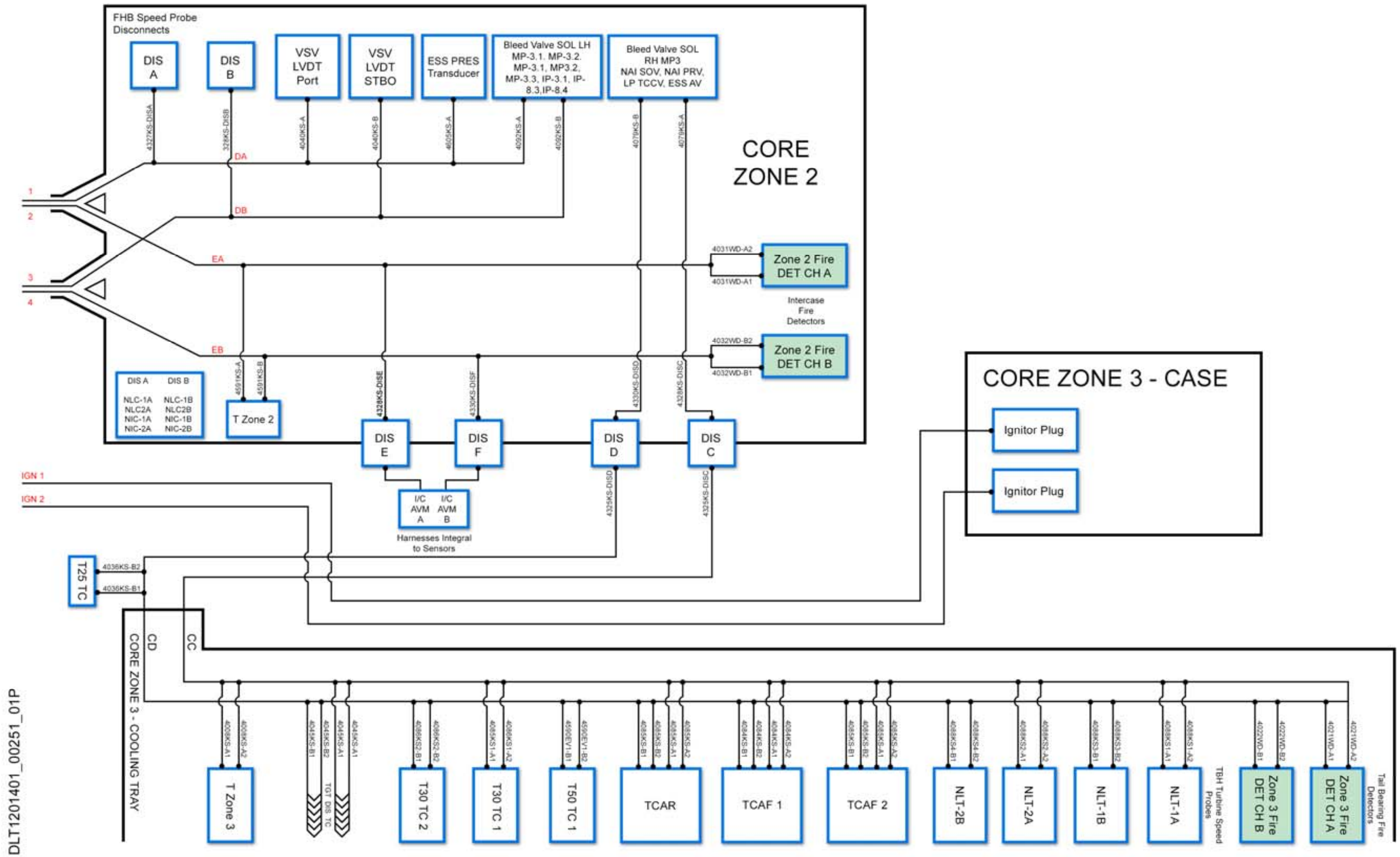

ZONES 2 & 3 WIRING DIAGRAMS

Section 11 – Engine Fire Protection Systems

Objectives

The student should now be able to:

- State the purpose of the Engine Fire Protection System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Fire Protection System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Engine Fire Protection System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Fire Protection System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Fire Protection System interfaces with other engine and aircraft systems.

End of Engine Fire Protection Systems

Section 12 - Engine Ice Protection System

Section 12 – Engine Ice Protection System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose of the Engine Ice Protection System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Ice Protection System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Engine Ice Protection System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Ice Protection System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Ice Protection System interfaces with other engine and aircraft systems.

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Engine Ice Protection System

Introduction

When the engine is operating in conditions of low temperatures and high humidity damage to the engine or loss of performance may take place due to the formation of ice.

Ice formation may occur on the leading edge of the inlet cowl and the spinner. Also, due to the large diameter fan, ice may form on the Engine Section Stator (ESS) vanes.

Ice forming on the two aircraft mounted ice detectors gives either; A.ICE ICE DETECTED or A.ICE SEVERE ICE DETECTED flight deck warnings on the Control and Display System (CDS). For back-up two forward fuselage mounted, illuminated indicators can be easily seen by the crew.

Location

The Trent XWB is equipped with anti-ice protection in the following areas:

- Nacelle Anti-Ice (Thermal).
- The Spinner (Dynamic).
- Core Anti Ice System – Engine Section Stators (Thermal).

Purpose

The purpose of the anti-icing systems is to prevent the build up of ice on the above areas when the engine is operating in low temperature and high humidity conditions.

Description

Nacelle Anti-Ice System

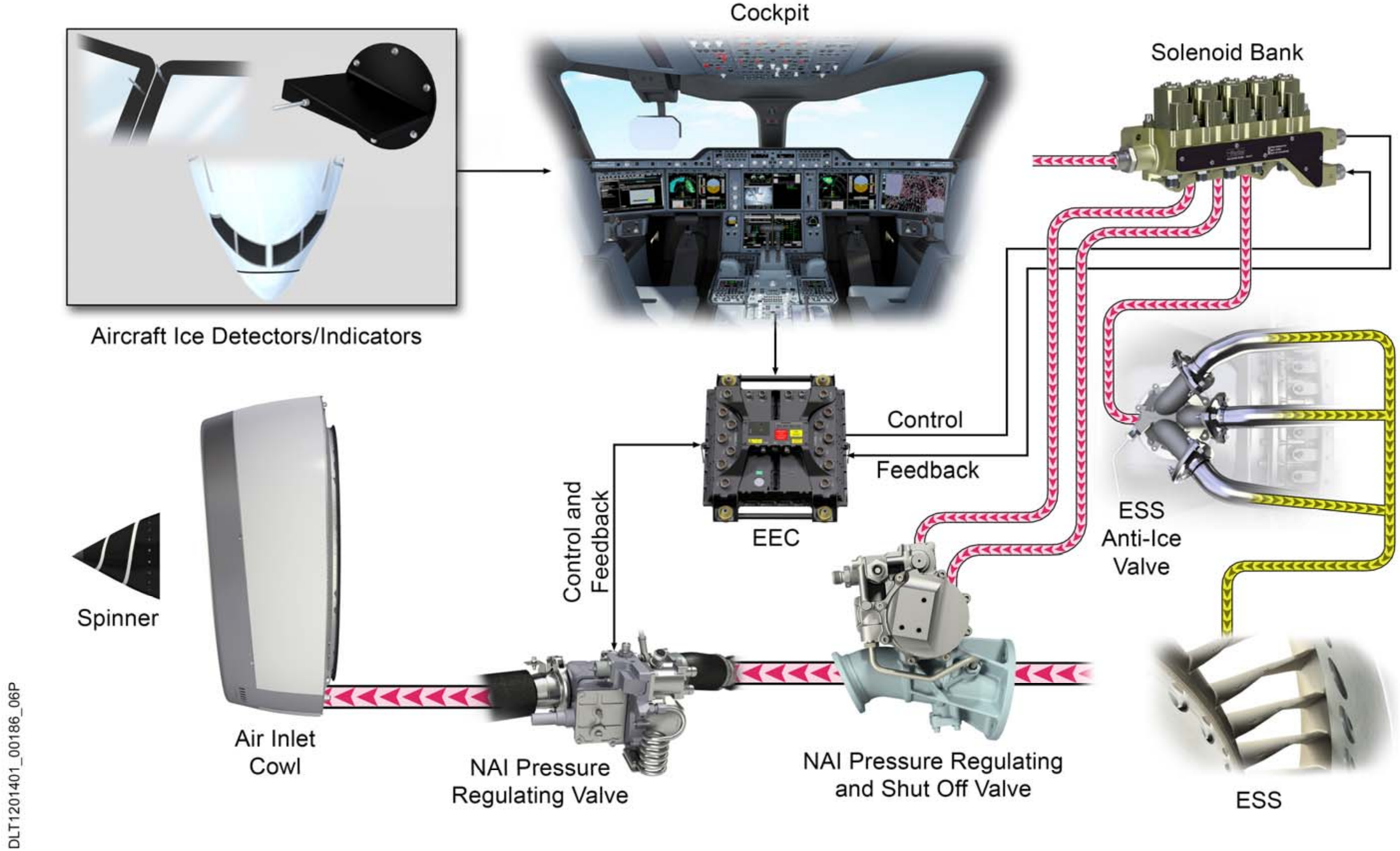
The nacelle leading edge protection is manually selected from the flight deck during icing conditions or when ice warnings are displayed. Selection will send an electrical signal to the RH solenoid bank and the Nacelle Anti-Ice Pressure Regulating Valve (NAI PRV) solenoid via the EEC, and so allowing HPC stage 3 servo air to operate the Nacelle Anti-Ice Pressure Regulating and Shut-Off Valve (NAI PRSOV) and activate NAI PRV, these valves will control the supply of hot air from the HP compressor stage 3 (HPC3) via ridged tubes to the D chamber cyclone ring swirling the air around the inner / front of the nacelle inlet cowl leading edge and exiting the air to atmosphere.

The Spinner

A solid flexible rubber cone is attached to the tip of the LP compressor spinner. When ice builds up on the tip, the tip vibrates and so shedding any ice build-up.

Engine Anti-Ice - Core Anti-Ice System

The EEC using various engine, aircraft and physical parameters automatically selects the core engine anti-ice protection, selection will send an electrical signal to the solenoid bank, and so allowing HPC stage 3 servo air to operate the Engine Section Stator Anti-Ice Valve (ESS AIV) and so supplying IP compressor stage 8 air via rigid tubes to the engine section stators at the front of the IP compressor. Core engine anti-ice protection will also operate when nacelle anti-ice is manually selected.



NACELLE ANTI-ICE SYSTEM

Nacelle Anti-Ice System

Nacelle Anti-Ice Pressure Regulating and Shut-Off Valve (NAI PRSOV)

Location

The NAI PRSOV is located on the lower left side of the HP compressor case.

Purpose

The purpose of the NAI PRSOV is to control the supply of HP3 air for nacelle anti-icing and act as a back-up to regulate the delivery of HPC stage 3 air to the internal D section of the leading edge.

Description

The valve body is bolted to the compressor case and is connected to the delivery duct by a V-band clamp.

The NAI PRSOV is a dual solenoid operated pneumatically controlled valve. HP3 servo air is supplied to the valve in two rigid tubes from the RH solenoid bank. Nacelle anti-ice is operated using cockpit controls through electrical harnesses via the EEC to the solenoids.

Failsafe; if electrical failure occurs both solenoids will de-energised and the valve will move to the fully open position.

Nacelle Anti-Ice Pressure Regulating Valve (NAI PRV)

Location

The NAI PRV is located on the lower left side of the LP compressor case.

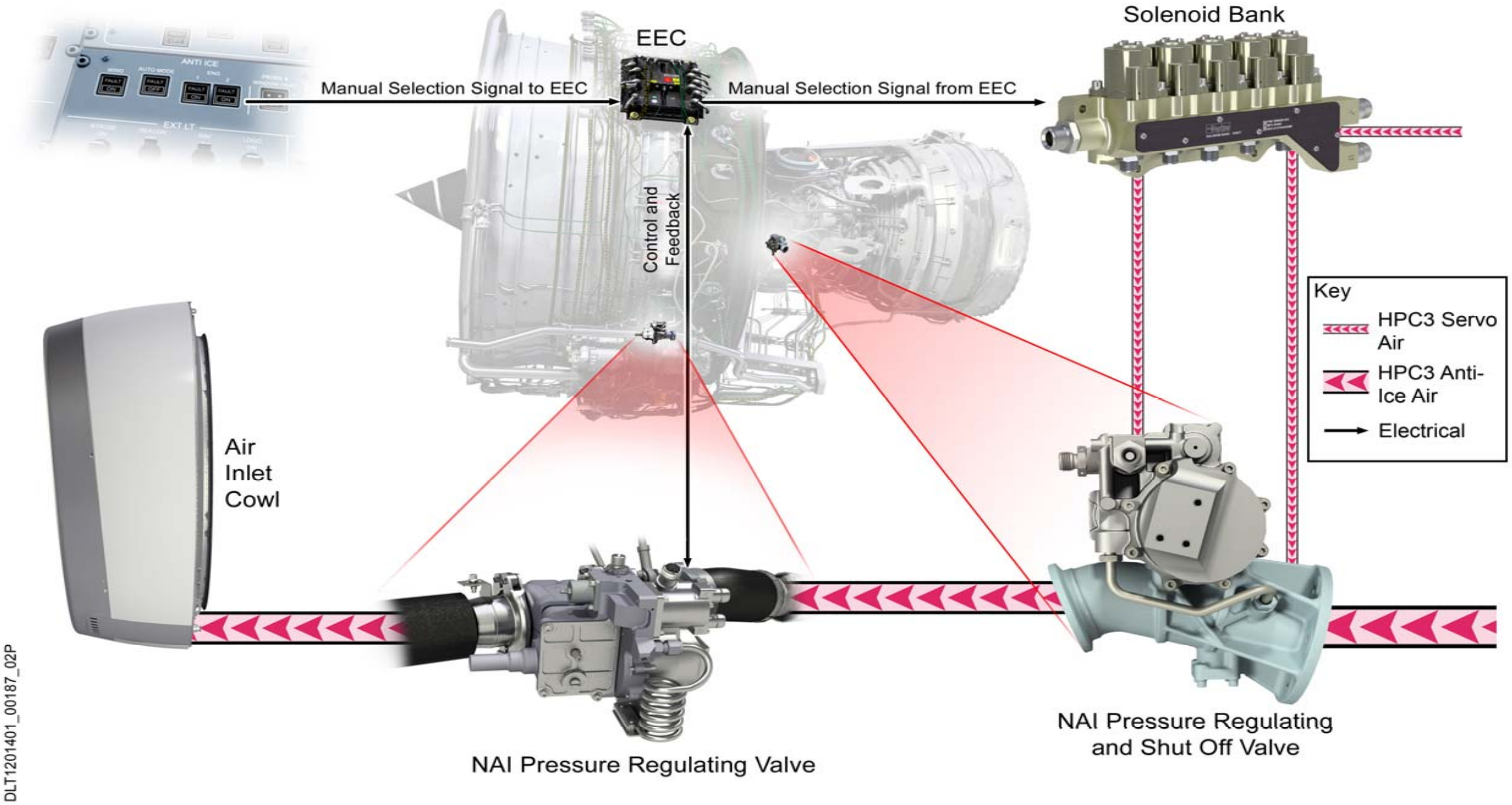
Purpose

The purpose of the NAI PRV is to regulate the delivery of HPC stage 3 air to the internal D section of the leading edge.

Description

The NAI PRV valve is connected to the anti-ice ducting using V-band clamps. It is a solenoid operated pneumatically controlled valve and is operated using cockpit controls through electrical harnesses.

Failsafe; if the pressure sensor signals a valve failure to the aircraft system the NAI PRSOV will be commanded to the regulating position. If electrical failure occurs the solenoid will de-energised and the valve will move to the regulating position.



NACELLE ANTI-ICE VALVE LOCATIONS

Nacelle Anti-Ice System

NAI PRSOV Functionality

- Will fully open when remotely selected on, from the flight deck, supplying HP3 air to the Nacelle Anti-Ice Ducting and Distribution Circuit.
- Operates as a redundant HP3 Regulating Valve when the Zone 1 NAI PRV is not operating or not able to operate.
- Regulates HP3 Anti-Icing Air to 75 +/-5 psig.
- Will fully close to restrict HP3 air from entering the Nacelle Anti-Ice Ducting and Distribution Circuit when not selected.
- Can be manually moved and locked in either the open or closed positions.

Major Components

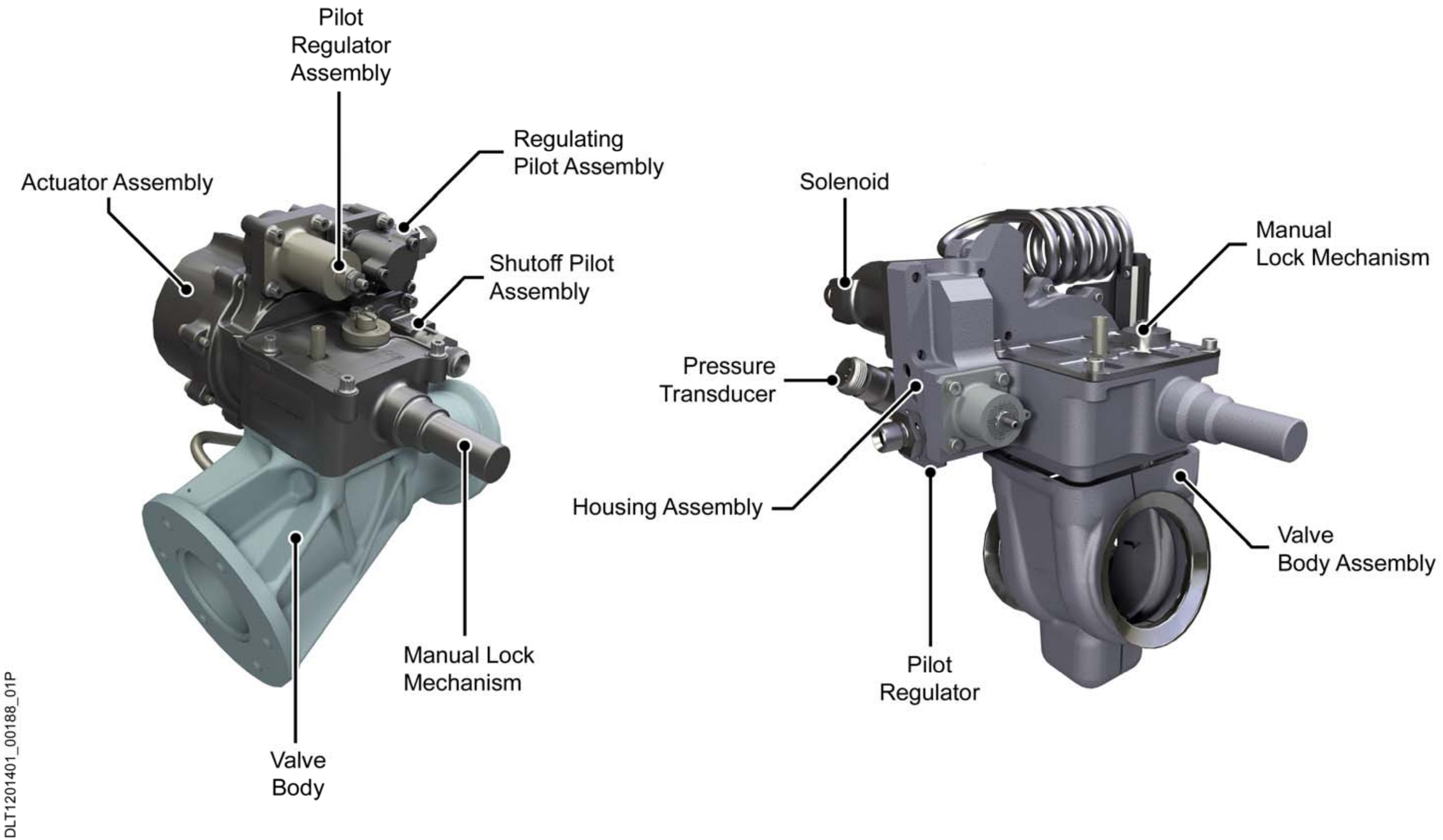
- Valve Body Assembly
- Actuator Assembly
- Pilot Regulator
- Shut-off Pilot Assembly
- Regulating Pilot Assembly
- Manual Lock Mechanism

NAI PRV Functionality

- Will go to regulating mode when remotely selected on from flight deck, supplying regulated HP3 air to the Nacelle Anti-Ice Ducting and Distribution Circuit.
- Has one Absolute Pressure Transducer which provides signals to indicate outlet pressure.
- Will fully open when not selected.
- Positioned downstream of the Zone 3 NAI PRSOV.
- Regulates HP3 Anti-Icing Air to 75 +/-5 psig.
- Can be manually moved and locked in the open position.

Major Components Include:

- Valve Body Assembly
- Actuator Housing Assembly
- Pilot Regulator
- Solenoid
- Pressure Transducer
- Manual Lock Mechanism



NACELLE ANTI-ICE VALVES

Nacelle Anti-Ice System

PRSOV & PRV regulating modes

The PRSOV & PRV have four regulating modes:

Mode 1. Selected - Pneumatic pressure less than regulation set pressure.

Mode 2. Normal operation.

Mode 3. Zone 3 in regulation mode.

Mode 4. Zone 3 shut off.

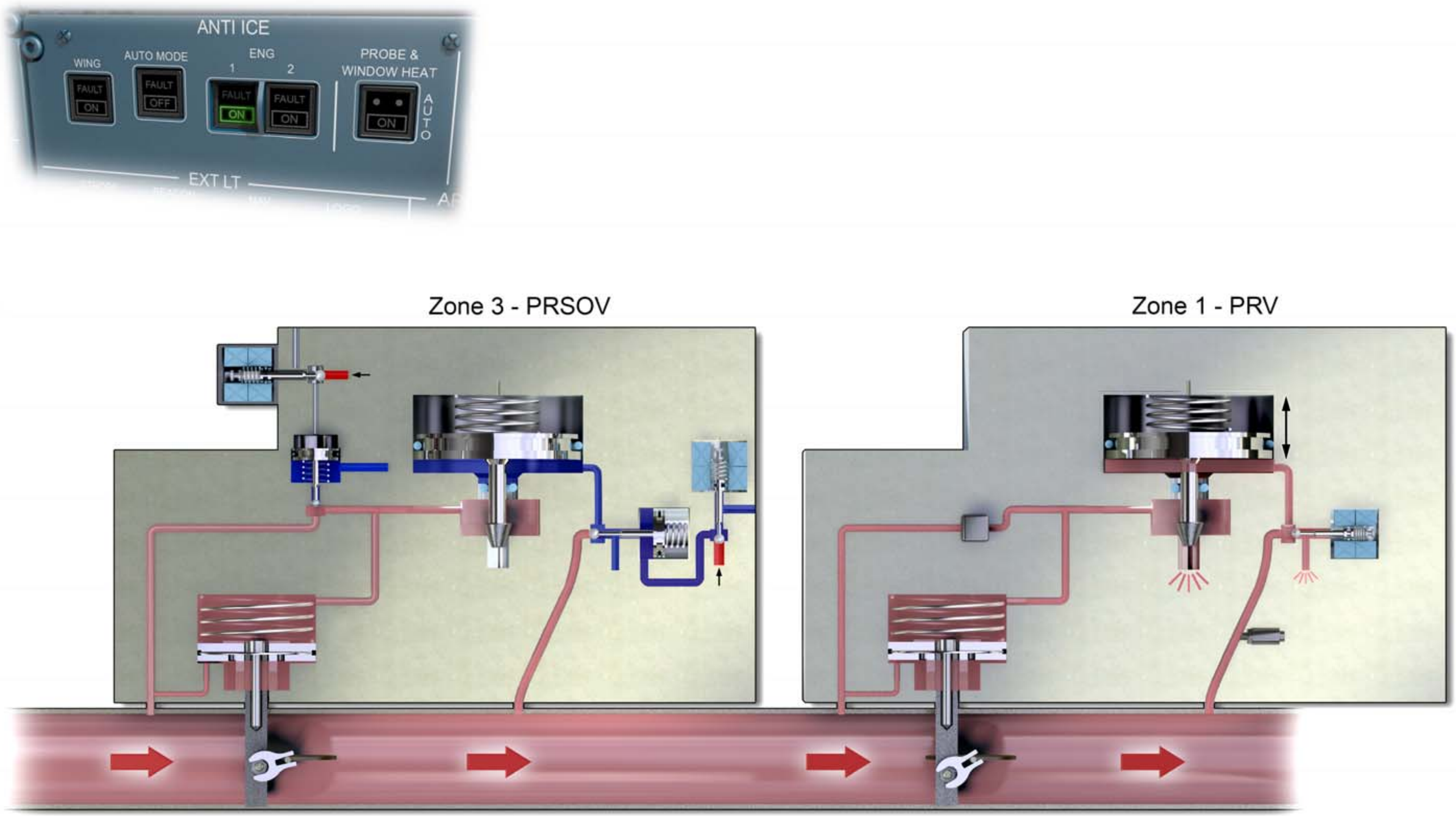
Mode 1 Selected - pneumatic pressure less than regulation set pressure.

PRSOV

- Inlet pressure less than regulation set pressure.
- Valve in full open position.
- Both solenoids De-Energised.

PRV

- Inlet pressure less than regulating set pressure.
- Solenoid De-Energised.
- Valve in regulating position, but fully open because pressure less than regulating pressure.



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NACELLE ANTI-ICE VALVES MODE 1

Nacelle Anti-Ice System

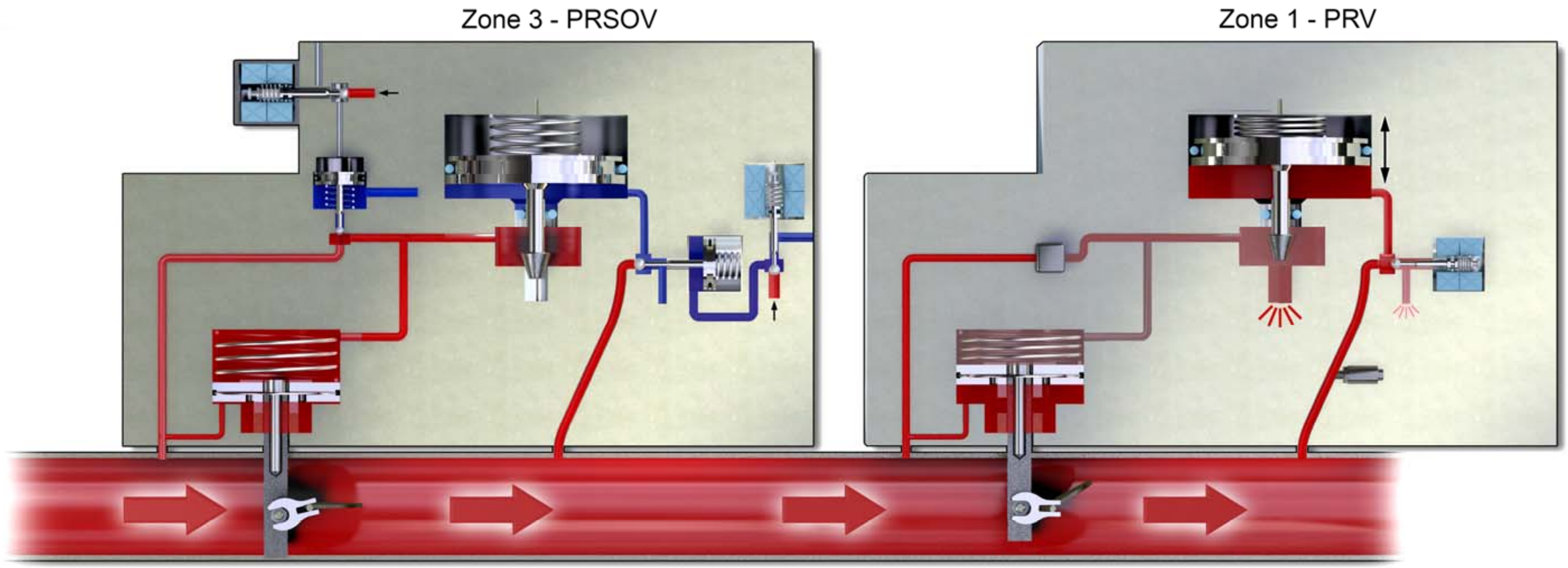
Mode 2 Normal operation

PRSOV

- Inlet pressure higher than regulation set pressure.
- Valve is full open.
- Both solenoids De-Energised.

PRV

- Valve in full regulation mode.
- Solenoid De-Energised.



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NACELLE ANTI-ICE VALVES MODE 2

Nacelle Anti-Ice System

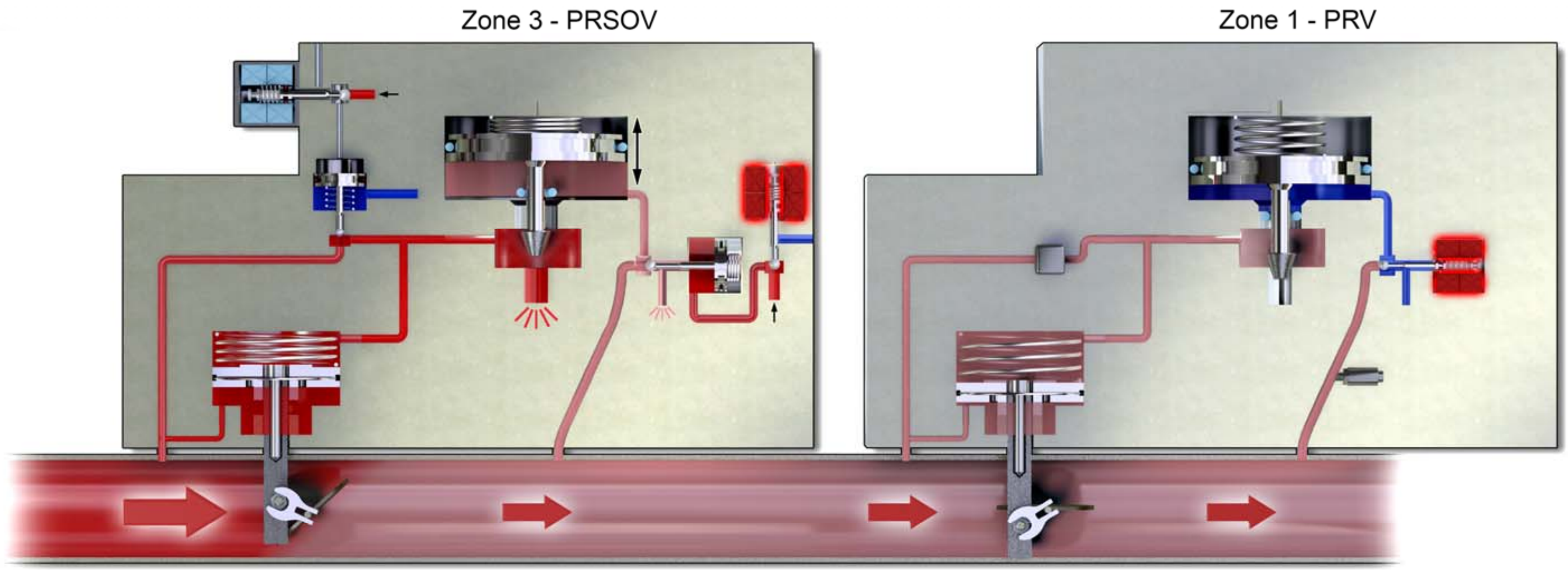
Mode 3 zone 3 in regulation mode

PRSOV

- Inlet pressure higher than the regulation set pressure.
- Regulating solenoid energised.
- Shut off solenoid De-Energised.
- Valve in full regulation mode.

PRV

- Zone 1 solenoid must be energised.
- Zone 1 regulator in full open mode.



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NACELLE ANTI-ICE VALVES MODE 3

Nacelle Anti-Ice System

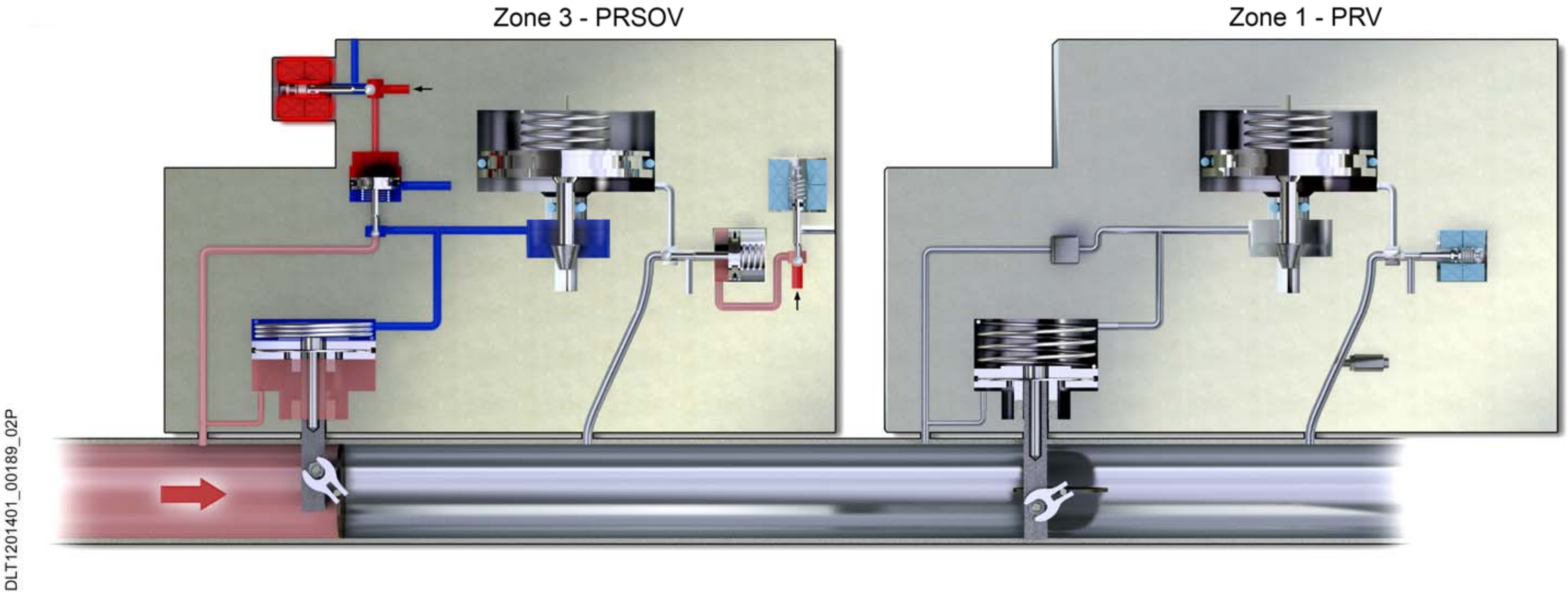
Mode 4 zone 3 shut off

PRSOV

- Inlet pressure higher than minimum operating pressure.
- Regulating solenoid De-Energised.
- Shut off solenoid energised.
- Regulator moves to the fully closed position.

PRV

- No shut off feature for zone 1 regulator.
- Regulating solenoid De-Energised.
- Valve in the regulating position.
- But full open because the pressure is < regulating pressure.



NACELLE ANTI-ICE VALVES MODE 4

Nacelle Anti-Ice System

NORMAL OPERATION

- ENG ANTI ICE selected to ON in the cockpit.
- A signal is sent to the EEC to operate both the NAI PRV & NAI PRSOV.
- The Shut-Off solenoid on the RH solenoid bank is de-energised this vents the servo line pressure and the PRSOV is sprung loaded to the open position.
- The solenoid on the PRV body positions the valve to regulating.
- The PRV then regulates the HP 3 air pressure from the PRSOV delivery duct to the inlet cowl ‘D’ section.

PRV FAILED OPERATION

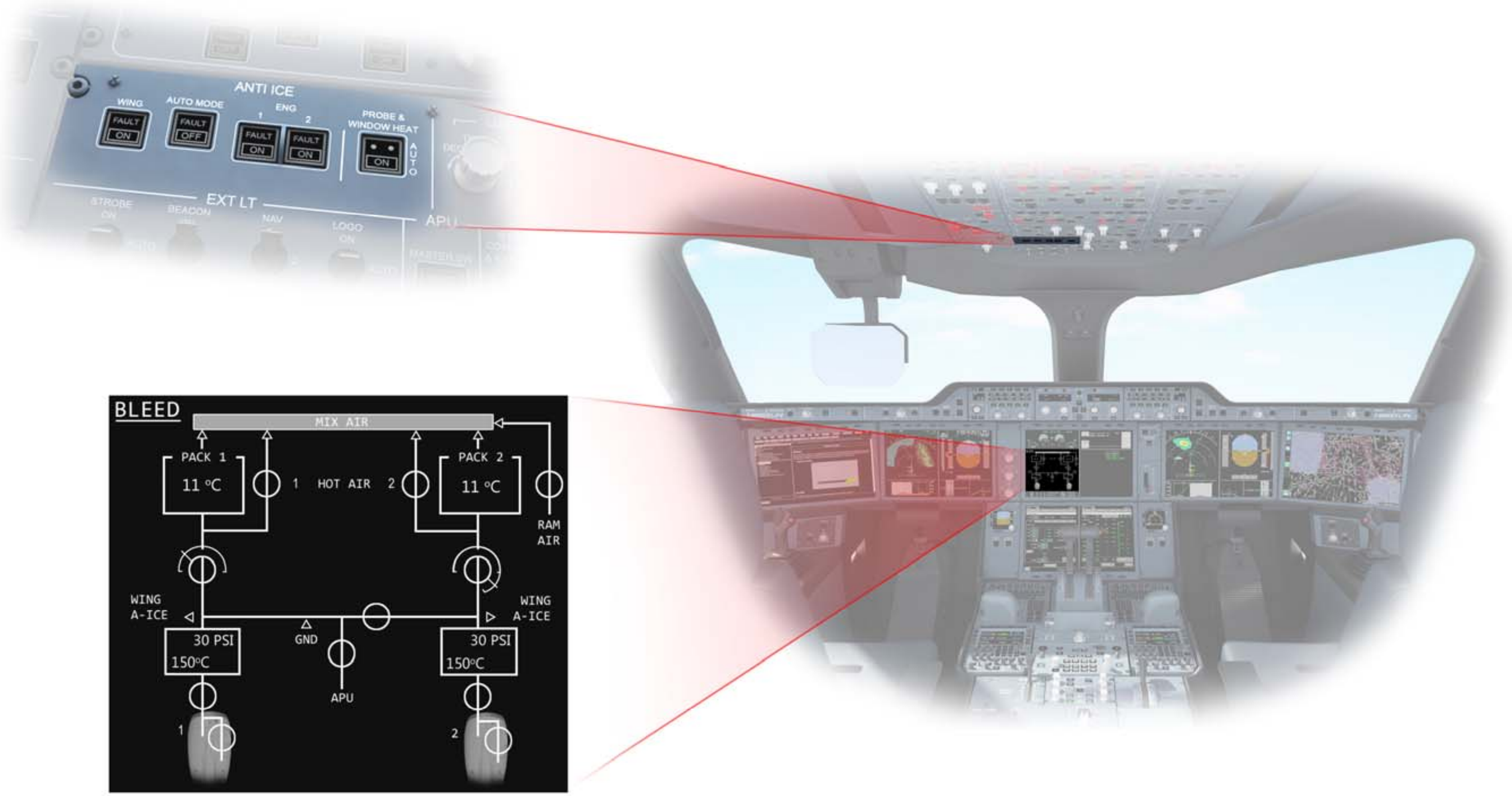
- Pneumatic failure the PRV fails to the regulating open position (sprung loaded) and a signal is sent to the EEC that the valve has failed (NOTE: - Electrical failure the PRV fails to the regulating position).
- The EEC then energises the regulation solenoid on the PRSOV.
- The PRSOV then regulates the HP3 air to 75± 5psi and supplies the delivery duct to the inlet cowl ‘D’ section.

TOTAL FAILURE IN FLIGHT

- The PRV fails to the regulating position
- The PRSOV fails to the open position
- Regulated HP3 air is sent to the inlet cowl ‘D’ section.
- A NAI anti ice fault signal is sent via the EEC to the cockpit.

FAILURE ON GROUND

- The PRSOV can be manually locked in the OPEN or CLOSED position for a limited ‘fly on’ condition. e.g. no icing conditions, limited altitude etc.
- The PRV can be locked in the OPEN position this allows the aircraft to operate for a limited amount of time before valve replacement, the PRV is left in position so the system is intact.



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NACELLE ANTI-ICE COCKPIT INDICATIONS / SWITCHING

Engine Anti-Ice - Core Anti-Ice System

Introduction

With cold and high moister ambient conditions and with the engine running at ground idle/low power, ice build-up could occur in the IP compressor. This Ice debris could cause mechanical damage to the IPC rotors. To prevent both the build-up of ice and reduce the size of ice shed from the ESS, heat conduction and warming of the air at the inlet to the IP compressor is use. To achieve this IP compressor Stage 8 air is ducted into the ESS vanes.

System Overview

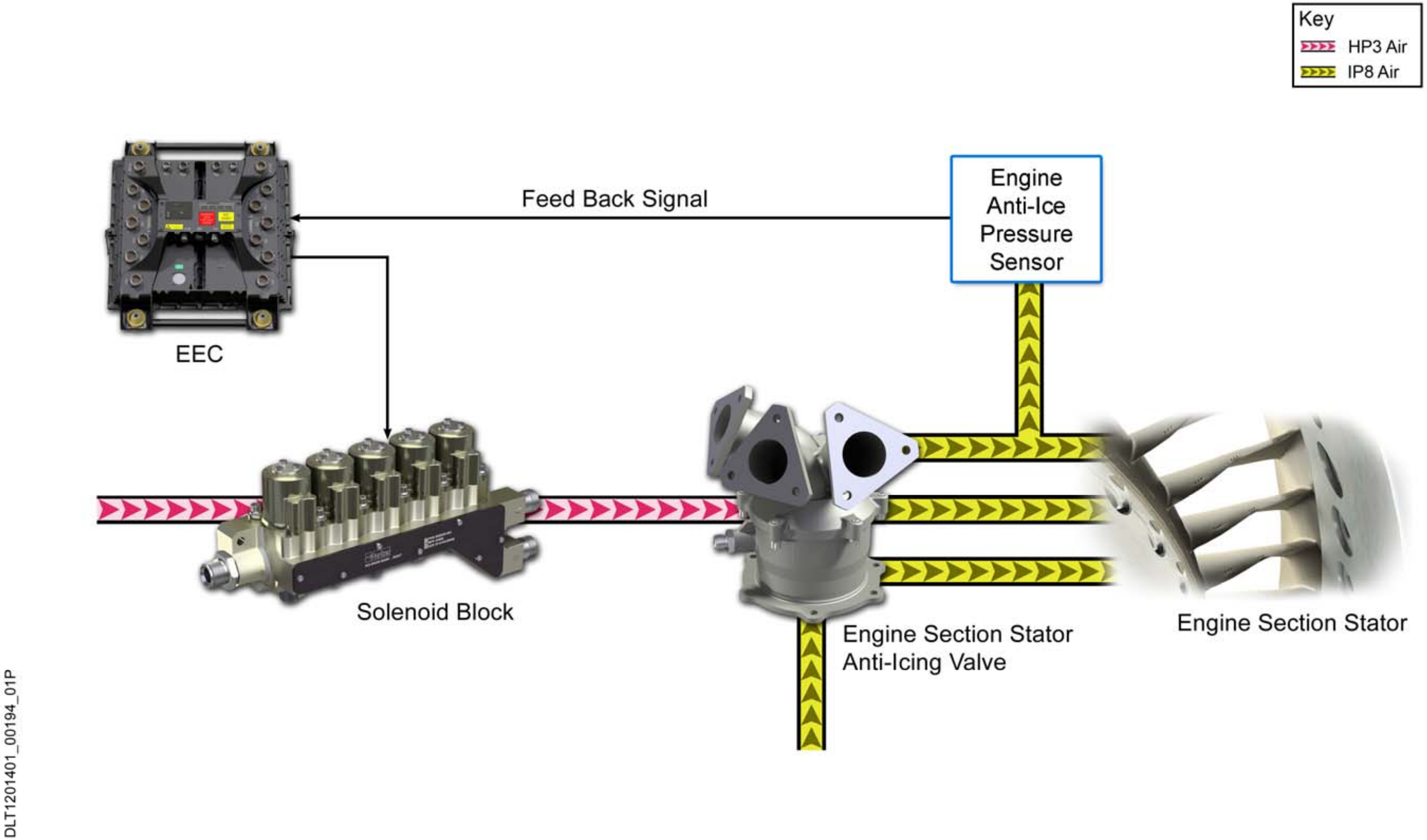
The core anti ice system consists of the following units:

- Electronic Engine Controller (EEC).
- An Engine Section Stator Anti-Ice Valve (ESS Valve).
- An Engine Anti-Ice Pressure Sensor.
- A dual channel ESS Solenoid valve.

The EEC

The EEC controls the ESS valve to the open or closed position via a solenoid as a function of T24, T0, N1, altitude, in-flight / ground status and nacelle anti-ice selection. It also provides a BITE function for the valve and solenoid, checking the systems serviceability.

The EEC controls the opening of the valve by sending a signal through electrical harnesses to energise or de-energise a solenoid in the right hand solenoid bank in zone 2. When the solenoid is energised HP3 servo air is supplied to the ESSAIV through a rigid tube and the valve will open against the spring force. IP8 air is then supplied to the engine section stators through three rigid tubes bolted to the valve outlet manifold. When the solenoid is de-energised the HP3 supply air is vented and the valve will be closed by spring force. A pressure transducer, located in one of the outlet tubes, monitors the downstream pressures to confirm the valve operation to the EEC.



ENGINE CORE ANTI-ICE SYSTEM

Core Anti-Ice System

Engine Section Stator Anti-Ice Valve (ESS AIV)

Location

The ESSAIV is located on top of the intermediate case in zone 2.

Purpose

To prevent the build-up of ice on the ESSs, the outer annulus of the IP Compressor Inlet and the VIGVs using IPC stage 8 air.

Description

The ESSAIV is bolted directly to the intermediate case at the 12 o'clock position and consists of a two position valve and a manifold with three outlet ducts. The anti-ice valve is opened by HPC stage 3 servo air via the solenoid bank and controlled by the EEC.

Engine Anti-Ice Pressure Sensor

Location

The anti-ice pressure sensor is mounted on the centre outlet pipe of the engine section stator anti-ice valve (ESSAIV).

Purpose

To indicate ESS Anti-icing failure, by detecting the ESSAIV outlet pressure.

Description

During both ground and in flight the EEC commands a BITE check (system ON and OFF) detecting a failure of valve operation by sensing the engine section stator anti-ice valve outlet pipe pressure, using an absolute simplex pressure sensor.

A dual channel ESS Solenoid

Location

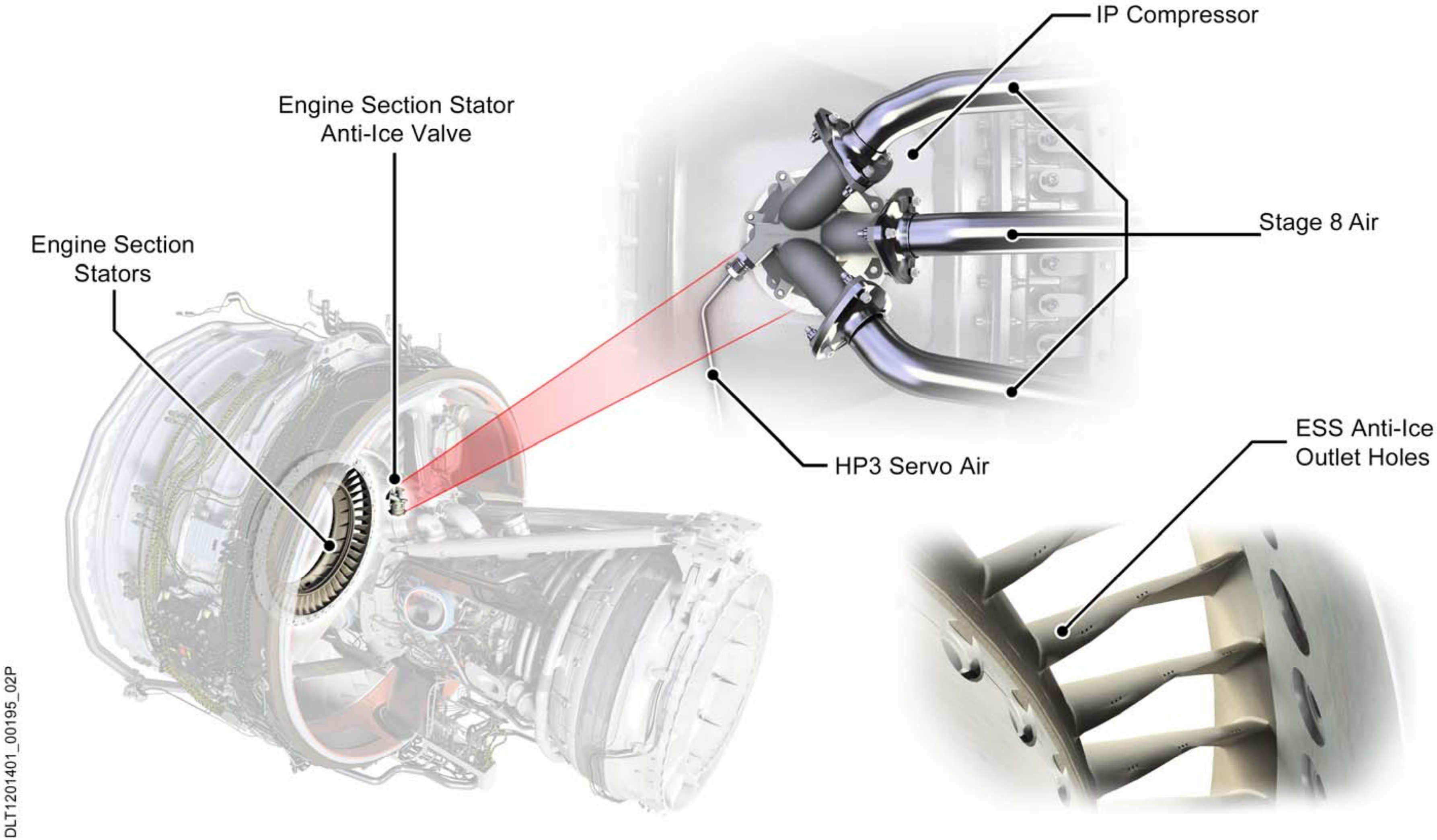
The solenoid is part of the right hand side solenoid bank situated on the top of the intermediate compressor case.

Purpose

The solenoid valve controls the flow of HPC stage 3 servo air to operate the opening and closing of the ESS valve when commanded by the EEC.

Description

Air is bled from HPC stage 3 to the solenoid bank and when electrically signalled by the EEC, the solenoid operates the valve in two positions open or closed when either energised or de-energised.



ENGINE SECTION STATOR ANTI-ICE VALVE

Core Anti-Ice System

Engine section stator anti-ice Normal Operation

The Engine Electronic Controller (EEC) transmits electronic signals to the RH solenoid block; the signal will energise or de-energise a solenoid. When the solenoid is energised HP3 servo air is supplied to the ESS Anti-icing valve through a rigid tube. The ESS Anti-icing valve is held in the closed position by spring force. HP3 air will push against the spring force and open the valve. Hot IP8 air is then supplied to the engine section stators and then released into the gas flow through openings in the stator. When the solenoid is de-energised the HP3 servo air is released to atmosphere and the valve is closed by spring force.

Engine section stator anti-ice Operation Restricting Factors.

The EEC will prevent the operation of the anti-ice system when the conditions that follow occur:

Ambient temperature less than - 30 °C

- This prevents operation when there is no risk of ice because of very low water content in the air.

T24 greater than + 10 °C

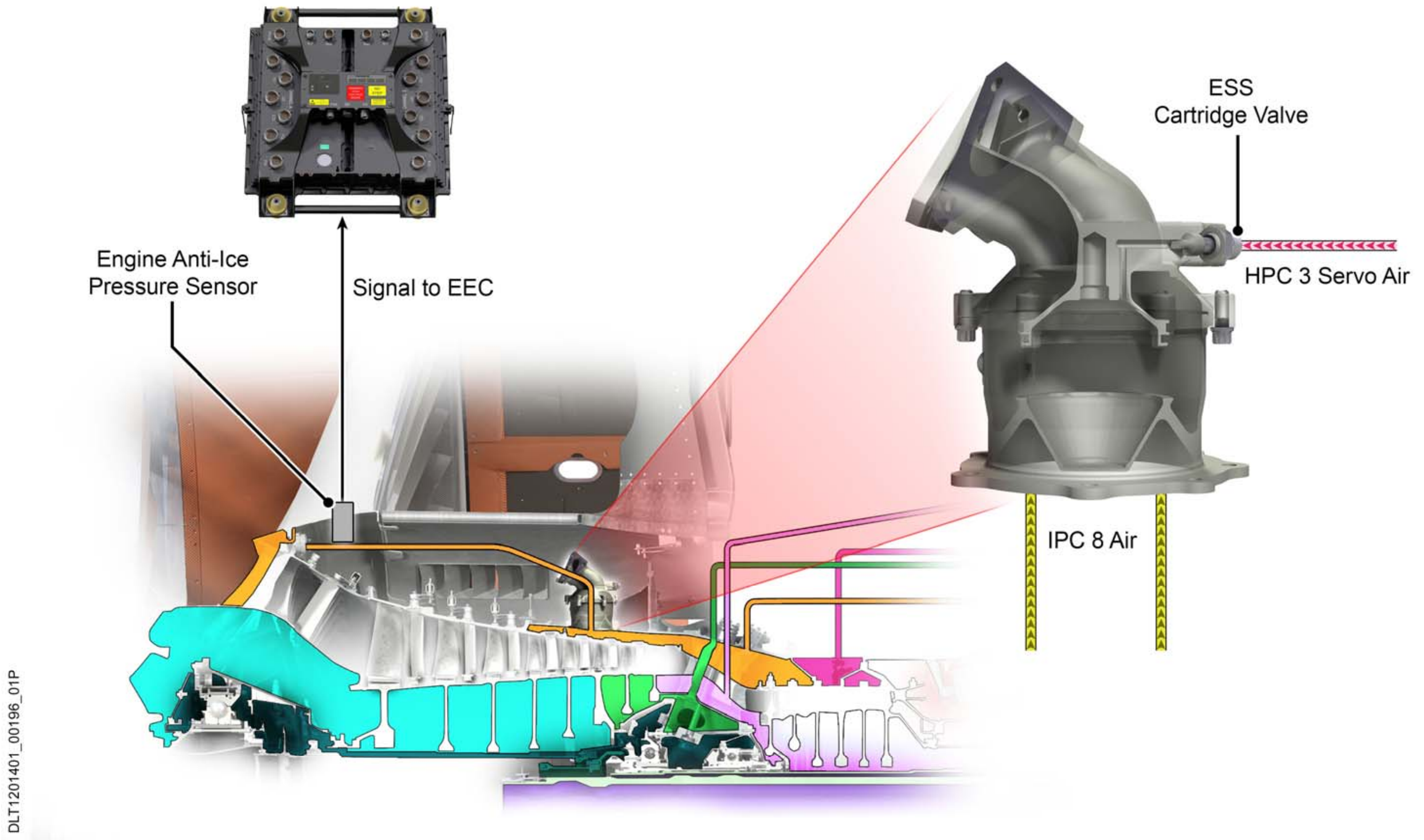
- This prevents operation when the total temperature at core entry is sufficient to prevent ice build up.

Altitude 31000ft

- This prevents operation at high altitude where clouds which contain super-cooled water are not usually present.

N1 greater than 65% on the ground or in latched take-off mode

- This prevents operation of the core anti-ice system during ground maintenance, engine running or in the take-off roll on cold days.

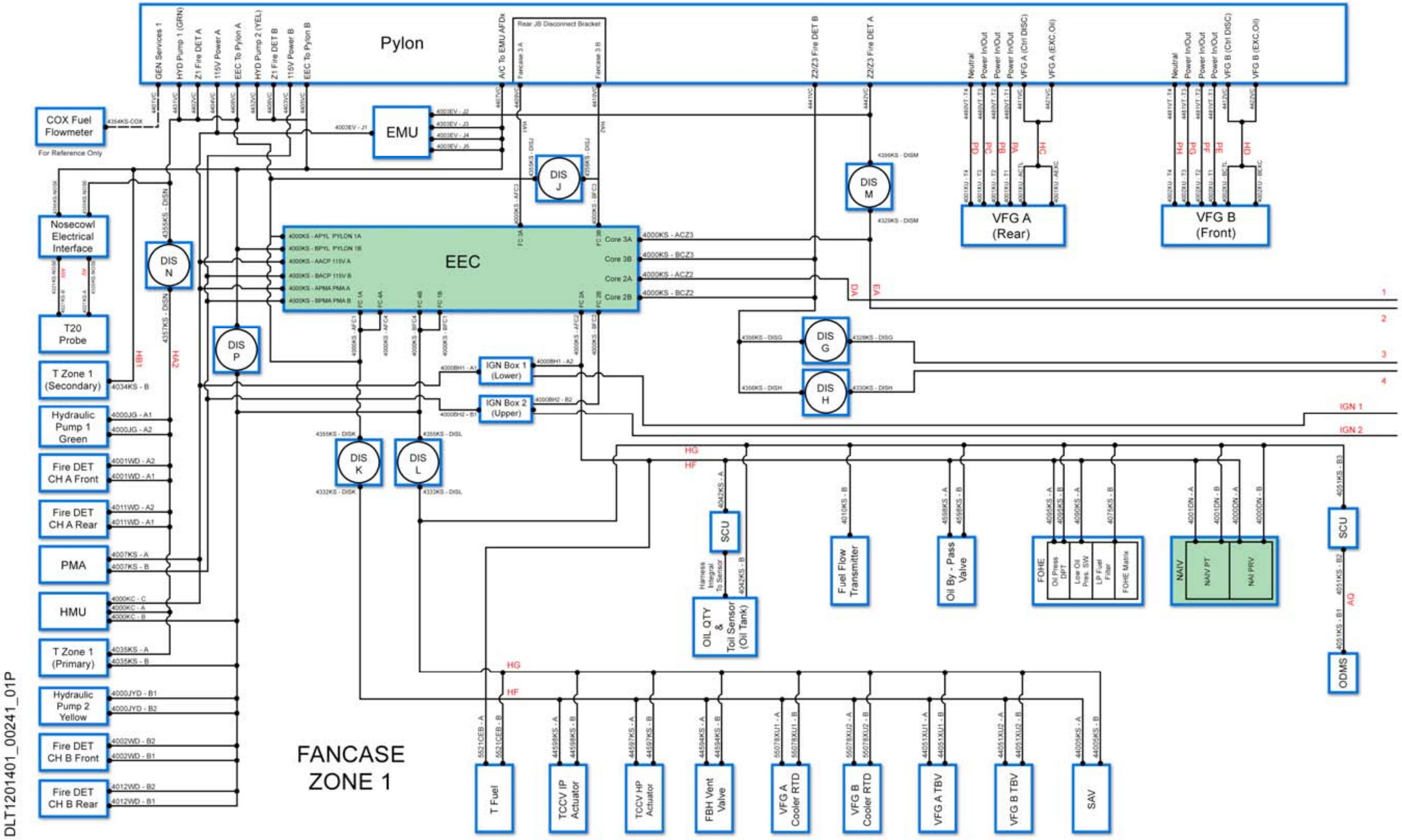


ESS ANTI-ICING SYSTEM OVERVIEW

Engine Ice Protection Systems Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.

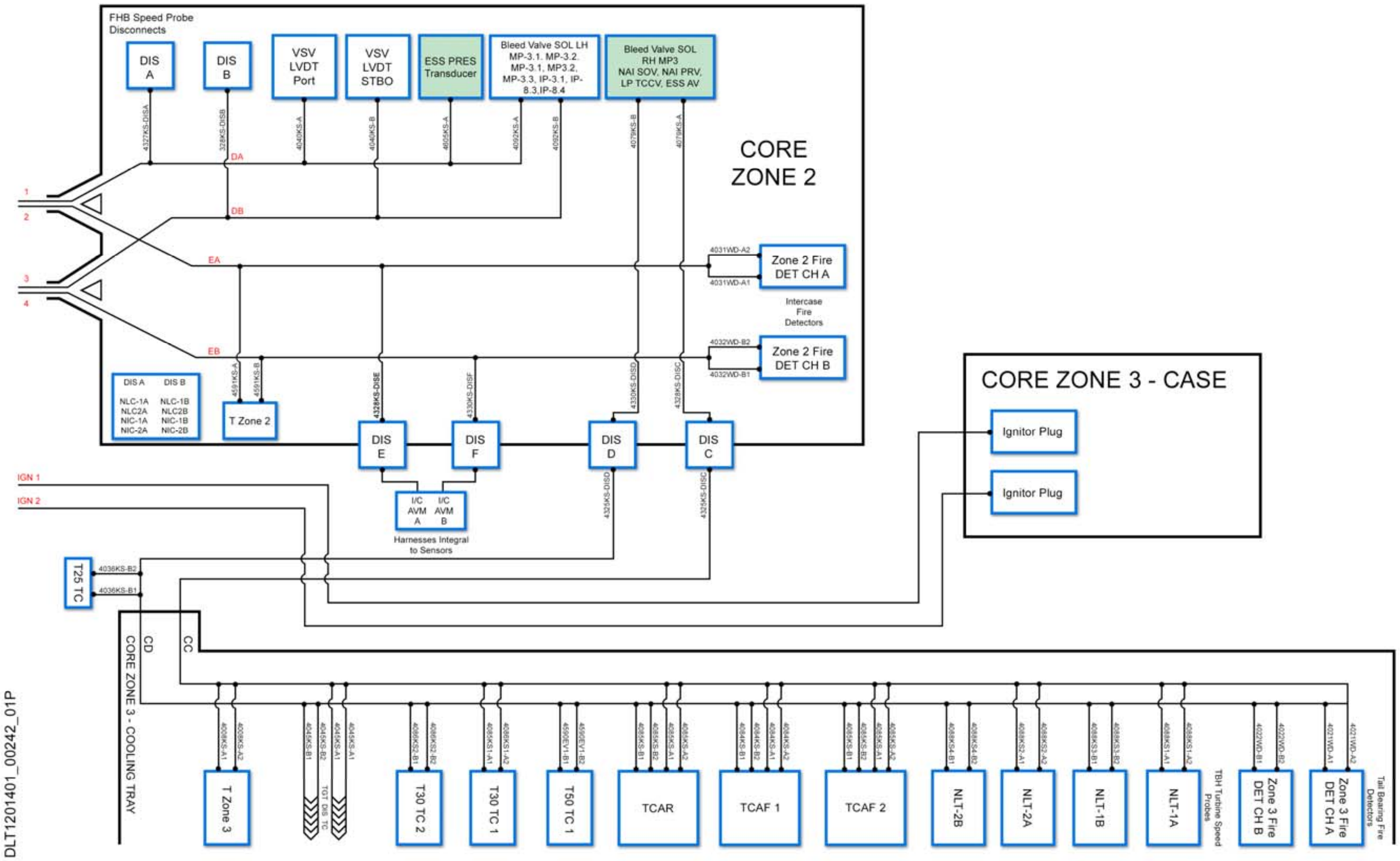


ZONE 1 WIRING DIAGRAM

Engine Ice Protection Systems Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.



ZONES 2 & 3 WIRING DIAGRAM

Section 12 - Engine Ice Protection System

Objectives

The student should now be able to:

- State the purpose of the Engine Ice Protection System as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the Engine Ice Protection System of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the Engine Ice Protection System of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the Engine Ice Protection System as fitted to the Trent XWB engine.
- Describe how the Trent XWB Engine Ice Protection System interfaces with other engine and aircraft systems.

End of Engine Anti-Ice System

Section 13 – Engine Starting & Ignition System

Section 13 – Engine Starting and Ignition System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

- State the purpose of the engine Starting and Ignition Systems as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Starting and Ignition Systems of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the engine starting and ignition systems of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the engine starting and ignition systems of the rent XWB engine.
- Describe how the Trent XWB engine Starting and Ignition Systems interface with other engine and aircraft systems.

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Starting and Ignition System Overview

Location

The starting and ignition system is located on the engine and within the aircraft.

Purpose

The purpose of the starting and ignition system is to enable rotation of the high-pressure system and the ignition of the fuel for usual engine ground starts and assisted in-flight starts.

The starting and ignition system is also used for on-ground maintenance tasks.

Engine Starting System

The engine starting system is pneumatically operated, and electrically controlled. The pressurised air required to operate the starting components is supplied from the following sources:

- The other running engine.
- The on-board Auxiliary Power Unit (APU).
- A ground based air compressor connected to the aircraft.

The pneumatically operated components are as follows:

- The Air Turbine Starter (ATS).
- The Starter Control Valve (SCV).
- Starter air ducting.

The engine can be started both on the ground or in-flight by switches in the cockpit, which operate the SCV via signals from the EEC.

Both channels of the EEC are connected the SCV solenoids via electrical harnesses, to signal the control of air flow to the ATS.

The ATS is attached to the external gearbox through which it rotates the HP system to a speed that will enable the engine to start.

Engine Ignition System

The engine ignition system is electrically powered from the aircraft electrical power supply and controlled by the EEC. The components of the ignition system are:

- Two High Energy Ignition Exciters.
- Two igniter leads.
- Two igniters.

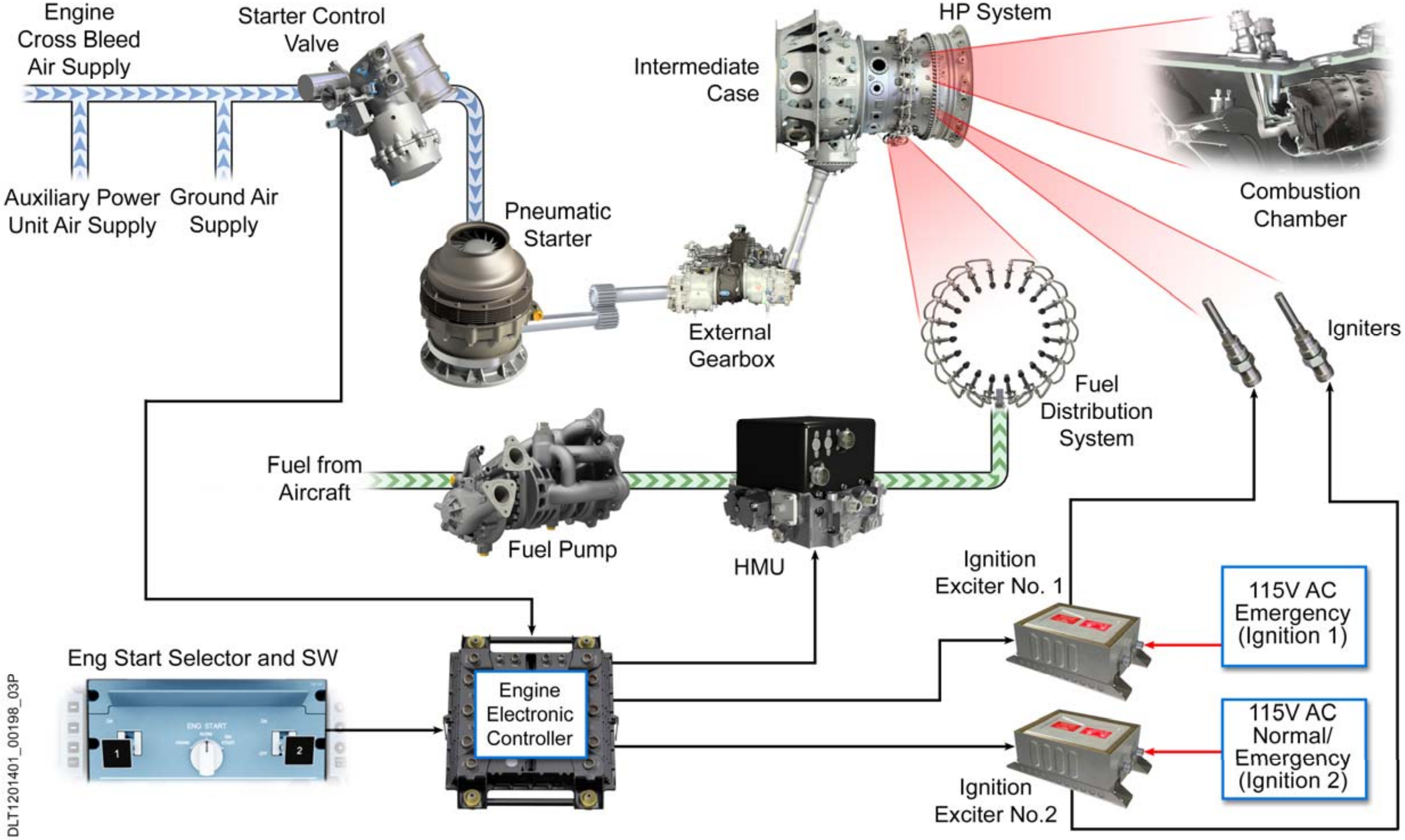
The ignition exciters are connected to each channel of the EEC by electrical harnesses.

The EEC controls the selection of the ignition system and alternates that selection on each start.

Engine Electronic Controller (EEC)

The EEC is used to control the starting and ignition system, controlling the starting sequences, engine cranking options and ignition selection in response to cockpit commands.

The EEC also interfaces with the HMU during the starting sequence to provide fuel to the combustion chamber via ridged pipelines and the primary and secondary manifolds to the fuel spray nozzles.



STARTING AND IGNITION SYSTEM OVERVIEW

Engine Starting – Cockpit Controls

Location

The engine starting controls are located on the centre pedestal and overhead panel.

Purpose

The purpose of the Cockpit controls is to allow the crew to select the mode of starting and cranking of the engine and initiate the starting / cranking sequence or engine shut down.

Description

Engine starting, shut down and cranking selections are made from the cockpit using the engine start selector, manual start switches and two engine master switches. Based on the selector and switch positions, the EEC initiates the starting or cranking operations.

The engine start selector, manual start switches and the engine master switch all interface with the EEC through the engine interface function (EIF) using the Avionics Full-Duplex Switched Ethernet (AFDX) network. For engine shut down, the crew operate engine master switch which is hardwired to the shut-off valve in the HMU.

Engine Start Selector

The engine start selector is located on the centre pedestal and has three positions (CRANK, NORMAL AND IGN/START). It allows for cranking and starting of the engine in combination with the engine master switches and manual start switches.

Manual Start Switches

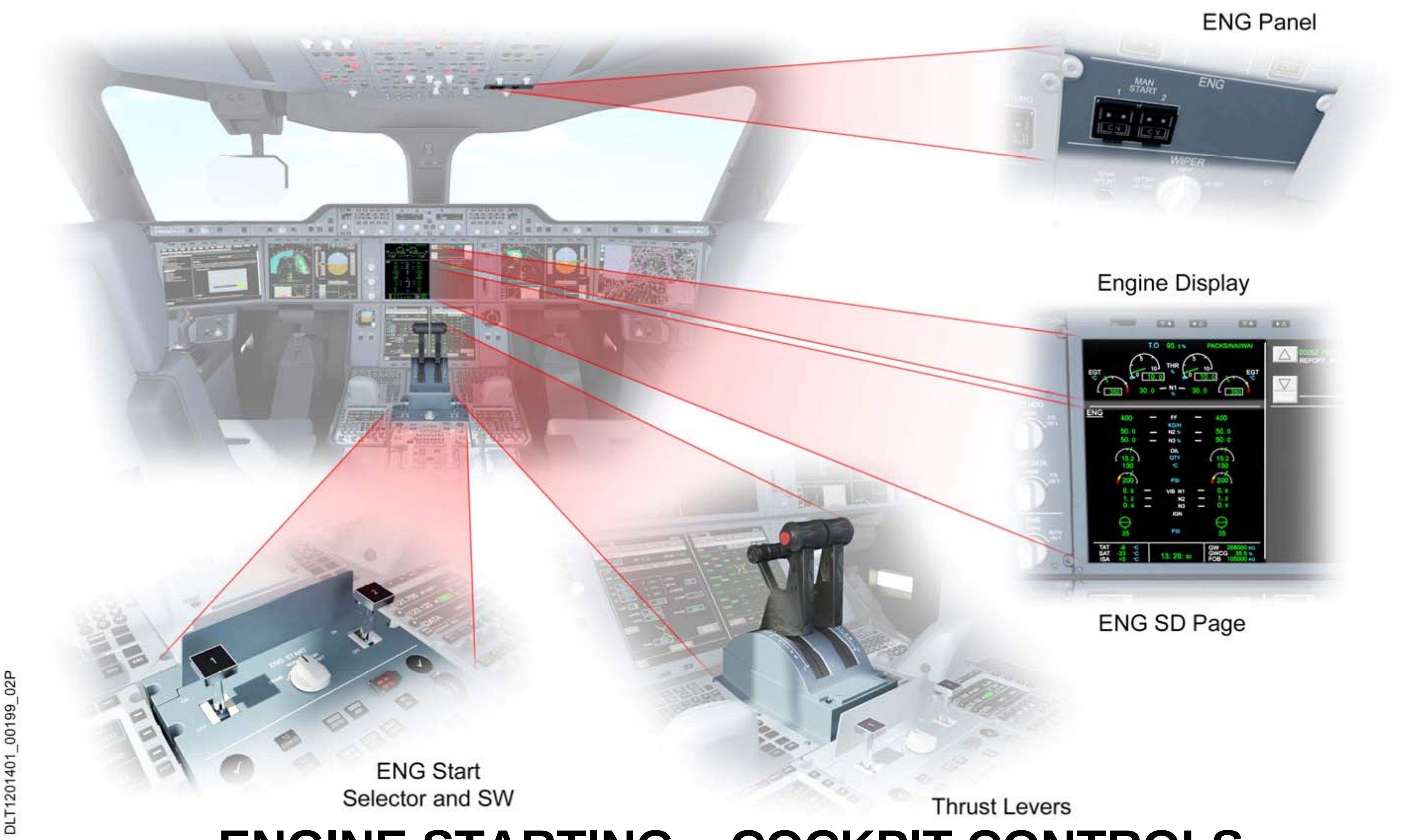
The two manual start switches, one for each engine, are located on the overhead panel. They allow the flightcrew to select a manual start in combination with the rotary switch.

Engine Master Switch

The two engine master switches are located on the centre pedestal are of the two position type ‘ON’ or ‘OFF’.

Selecting the master lever ‘ON’ will start/crank each engine in combination with the related engine start selector.

Selecting the master lever ‘OFF’ will shut the engine down.



ENGINE STARTING – COCKPIT CONTROLS

Cockpit Starting Indications

Location

The engine starting indications are located on the ECAM screen.

Purpose

The purpose of the starting indications is to provide the crew with a means to monitor the engine parameters during engine start.

Description

The starting indications are classified as secondary parameters; as such they are shown on the lower ECAM screen and are only displayed during the engine start sequence as follows:

- **Starter Control Valve (SCV**) – A symbol is shown to the crew to represent the SCV position during the engine start sequence and is signalled by the EEC. This symbol will also have a white box around it to indicate which engine is has been selected to start.
- **Igniter System (IGN)** – A digital indication shows the crew which ignition system the EEC is using during the engine start sequence. The letter A, B or AB defines the system used, **A** will be displayed to the left of the SCV symbol and **B** will be displayed to the right of the SCV Symbol and **A** and **B** will be displayed if the EEC selects both igniters. E.g. During a manual start.
- **Starter Air Duct Pressure** – A digital indication shows the crew the air pressure within the starter air ducting in pounds per square inch (PSI). This is displayed below the SCV symbol.

Ignitor plug / system identification

Exciter box 1 is connected to plug for channel ‘A’

Exciter box 2 is connected to plug for channel ‘B’

Note: Strictly follow the approved maintenance procedures when working on the ignition system. Dangerous electrical discharges can occur. Before working on the ignition system, isolate power supply and wait for a sufficient time.

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ENGINE STARTING – COCKPIT INDICATIONS

Engine Starter System

Starter Control Valve (SCV)

Location

The SCV is located around the 7 o'clock position in the starter air ducting on the left side of the LP compressor case.

Purpose

The purpose of the starter control valve is to provide control of pressurized air in the starter air duct to the Air Turbine Starter (ATS)

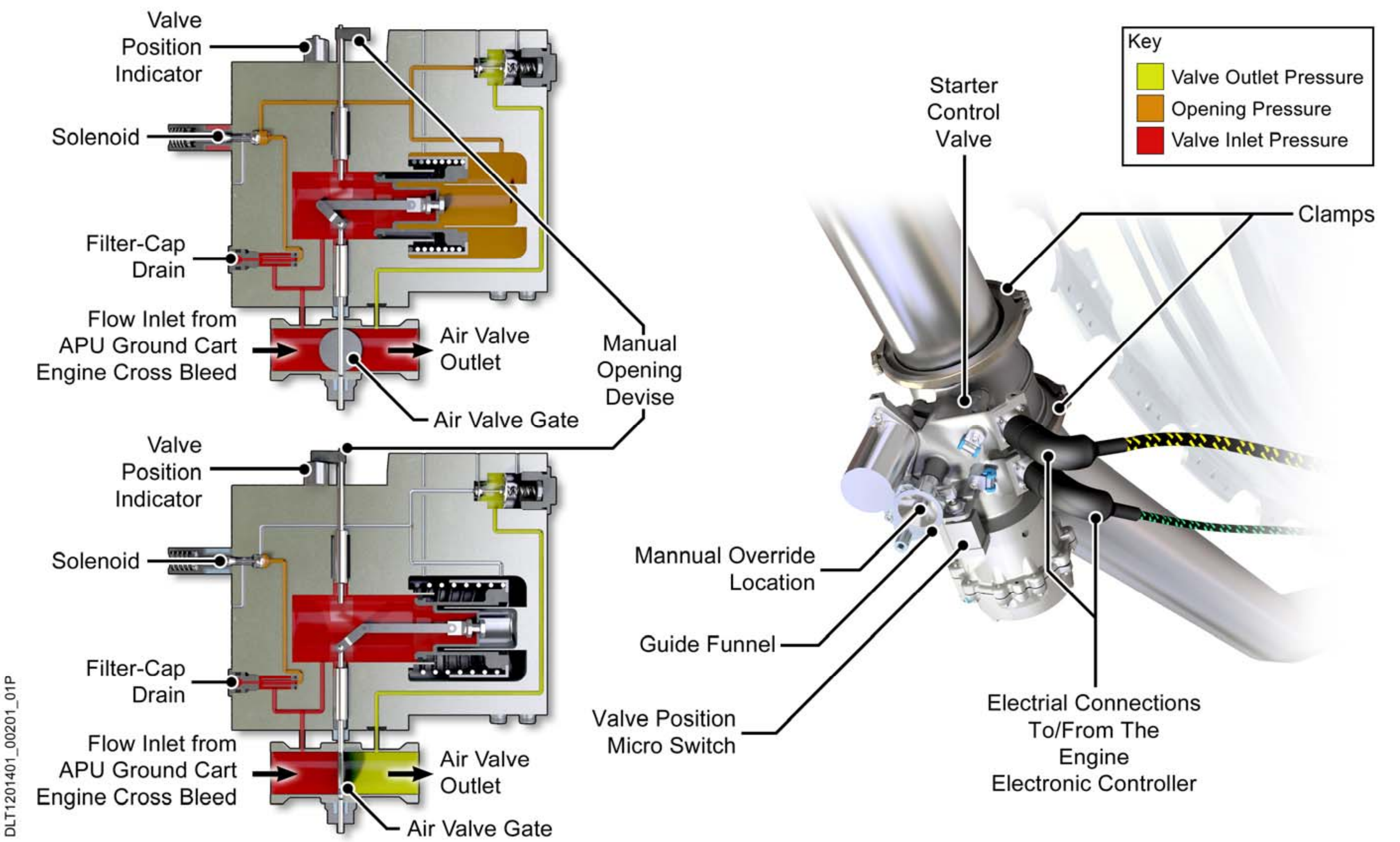
Description

The EEC, through an electrical harness, is connected to a solenoid that electrically controls the SCV. Air is supplied to the valve from the upstream section of the air ducting to operate the SCV.

The EEC commands the SCV to open using commands from the engine master or manual start switches in the cockpit. When open the pressurized air in the starter air ducting is directed to the ATS that rotates the engine HP system through the external gearbox, angle driveshaft and intermediate gearbox.

As the engine progresses through its start cycle the EEC will command the starter control valve to close when the ATS reaches its cutout speed. Feedback to the EEC of the valve position is provided by a micro switch.

On the outside of the valve is a visual valve position indicator and a manual opening device. Should the valve electrically fail, the ground engineer can open the SCV manually, allowing pressurized air to flow to the ATS and so start the engine turning. Cockpit master start switch selected to 'ON' will initiate the rest of the starting sequence.



STARTER CONTROL VALVE

Air Turbine Starter (ATS)

Location

The ATS is located on the left hand side, front face, of the external gearbox.

Purpose

The purpose of the ATS is to provide a rotational force to the HP system for the starting of engine and for engine cranking ground maintenance.

Description

The ATS is attached to the external gearbox using a quick attach detach (QAD) clamp.

Pressurised air from either the aircraft Auxiliary Power Unit (APU) or Ground Support Equipment (GSE) or the other operating engine is used to rotate an integral turbine. The turbine is connected to reduction gears which in turn are connected to an output shaft that is located into the external gearbox by splines.

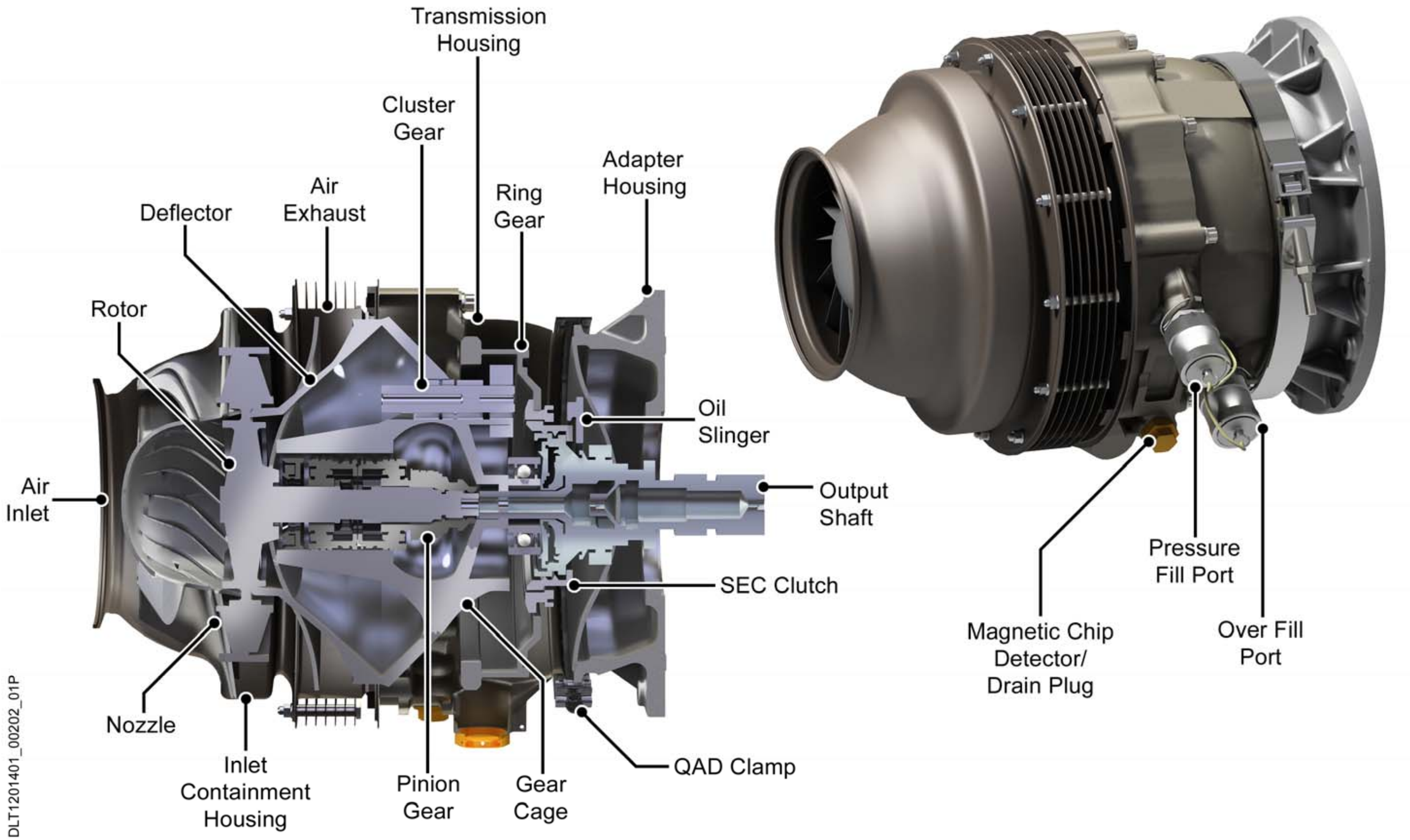
This output shaft transfers the high speed of the turbine into a high torque low speed rotational force. The driving force from the ATS rotates the HP system though the external gearbox, angle drive shaft and intermediate gearbox.

To prevent the ATS from turning the engine at a speed more than the limit, a clutch will disengage the drive from the turbine to the reduction gears at a pre-set HP speed. This is known as the starter cut out speed. The EEC will also close the Starter Control Valve (SCV) when the HP system is at the starter cut out speed.

The EEC is aware of the starter cut out speed as a function of its software and will close the SCV to stop the air flow to the ATS turbine.

The ATS has a self-contained oil system that has the following maintenance components:

- Pressures fill point – Provides a method of replenishing the oil.
- Over flow point – Provides a method of ensuring the oil is replenished to the correct level.
- Drain Plug – Provides a method of draining the oil.



AIR TURBINE STARTER

Engine Ignition System

Location

The engine ignition system is located on the LP compressor case and the combustor case.

Purpose

The function of the ignition system is to provide an electrical spark to ignite the fuel / air mixture in the combustion chamber to start the engine on the ground and in-flight. It is also used to ensure continued combustion during any flameout conditions and is divided into the following sub systems;

- The Electrical Power Supply System

  The function of the electrical power supply system is to supply electrical power to the igniter plugs.

- The Distribution System

  The function of the distribution system is to transmit electrical power from the ignition exciters to the igniter plugs.

- The Switching System

  The function of the switching system is to provide control of the ignition system through the EEC during engine start and relight operation.

Description

Each engine has a dual ignition system. One system is controlled by the Engine Electronic Controller (EEC) channel A and the other system is controlled by EEC channel B. These systems can operate together or independently. Each system has the following components:

- High Energy Ignition exciter.
- Igniter lead.
- Igniter plug.

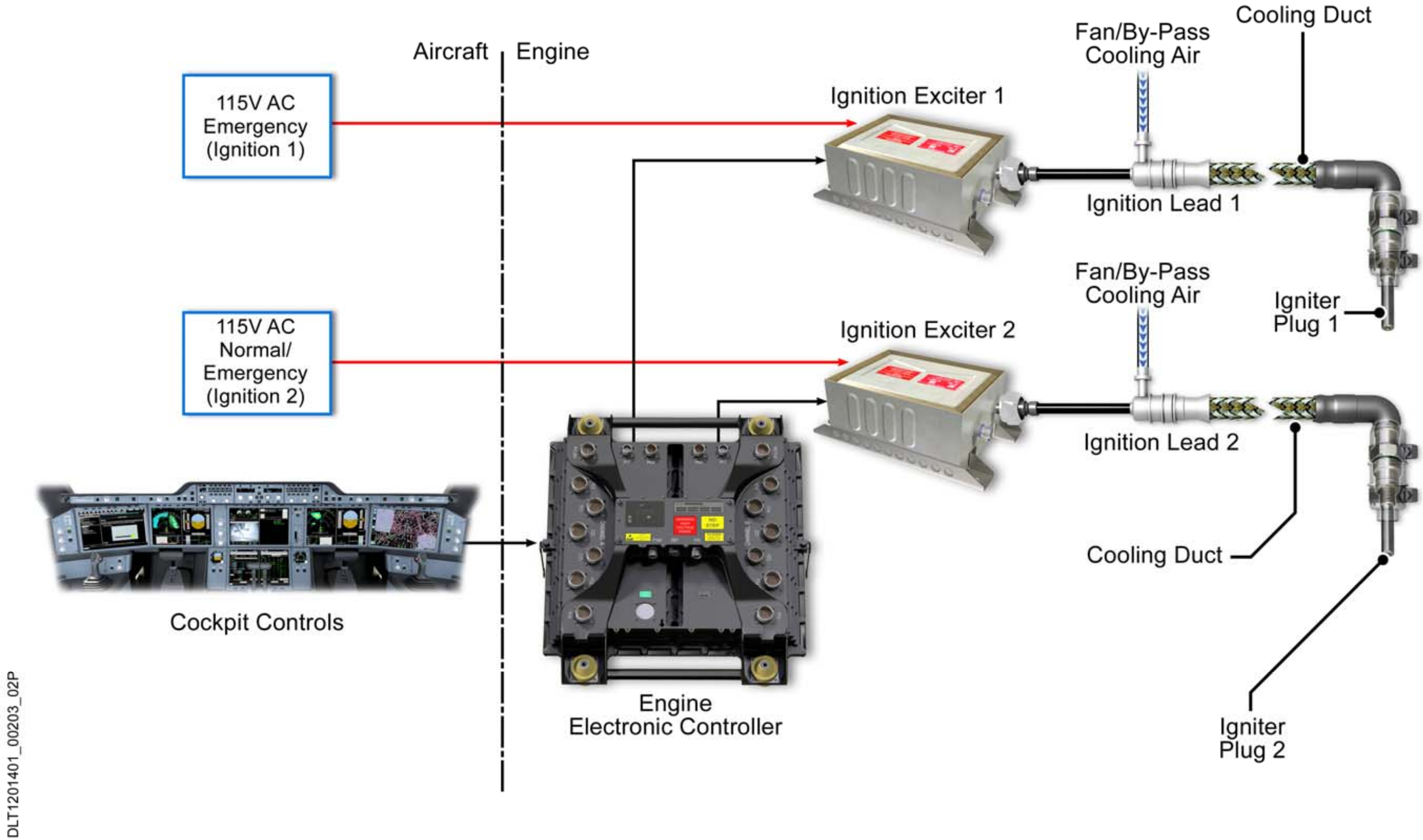
An electrical power supply is provided directly to the ignition exciter from the aircraft. The ignition exciter supplies a low-tension output voltage, at high energy levels, through the ignition leads to the igniter plugs.

In usual conditions the EEC commands the exciters to operate alternatively on each start i.e. system A on the first start of the day, system B on the second, system A on the third etc.

Should there be an abnormal condition which threatens to cause a flame out; the EEC will command both ignition systems to operate in an attempt to keep the fuel / air mixture alight.

The high-energy output of the ignition exciter is given to the igniter plug through an igniter lead. Due to the high temperatures surrounding the combustion case the igniter is partially cooled with air from the IP TCC valve at its lower section nearest the igniter plug.

The igniter plug protrudes into the combustion chamber and receives the electrical energy from the ignition exciter. The igniter plug will discharge the energy to produce a spark across its surface that ignites the fuel / air mixture within the combustion chamber.



ENGINE IGNITION SYSTEM

Ignition System Components

High Energy Ignition Exciters

Location

The two ignition exciters are located on the lower left hand side of the LP compressor case. The upper exciter No. 2 is channel ‘B’ and the lower exciter No.1 is channel ‘A’.

Purpose

The purpose of the ignition exciter is to provide a high-energy electrical supply to the igniter.

Description

Each ignition exciter is supplied with an independent AC electrical supply from the aircraft supplying a low-tension output voltage, at high energy levels. The EEC controls the selection of either or both ignition exciters. A capacitor within the ignition exciters will output a high-energy charge to the igniter plug when commanded by the EEC. Each engine has two ignition systems. Both systems are controlled by the EEC channel A & B and can operate together or independently.

Igniter Lead

Location

The two igniter leads are located between the ignition exciters and the igniter plugs.

Purpose

The purpose of the igniter lead is to transfer the high-energy supply from the ignition exciter to the igniter plugs.

Description

The ignition leads are approximately 18 feet long and have an inner core and protective outer case which, if required, can be replaced once removed from the engine. The ignition leads are attached at clipping points to the LP compressor case, the bifurcation panel and the combustion outer case. The lead connects to the igniter plug by a screw thread connection. Due to the high temperatures around the combustion system the leads are cooled by air at the section nearest to the igniter plug. The cooling air is supplied from the IP TCC valve inlets.

Igniter Plug

Location

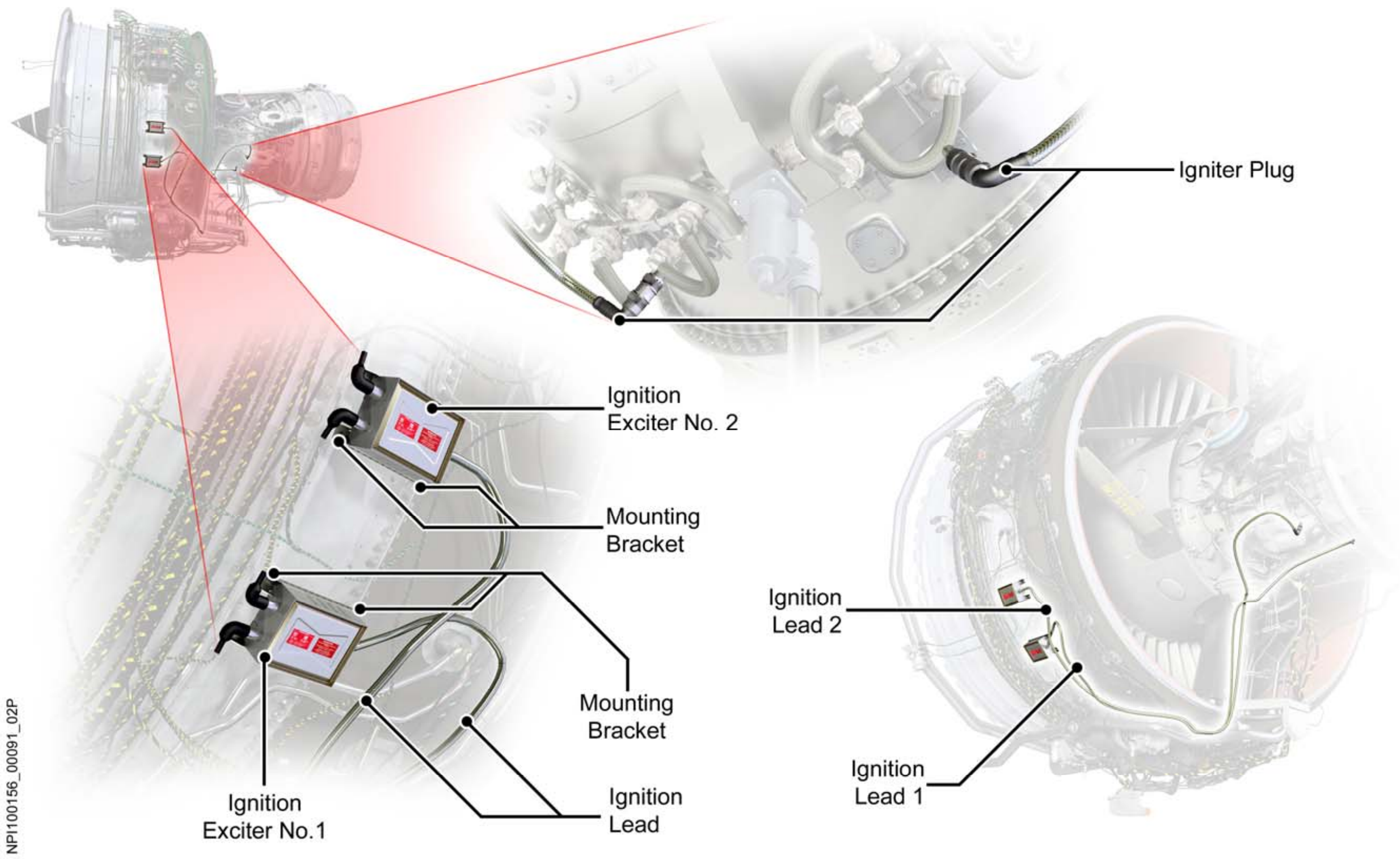
The two igniter plugs are located on the lower half of the combustion case. Channel ‘A’ plug is at the rear of fuel spray nozzle 12, and channel ‘B’ is to the rear of fuel spray nozzle 8.

Purpose

The purpose of the igniter plug is to ignite the fuel air mixture in the combustion chamber during starting / relight and also prevents an engine flame-out in bad weather conditions.

Description

The igniter plugs are installed into the lower half of the combustion outer case and attached to the ignition leads and protrude through the combustion case into the combustion chamber. The high-energy electrical supply is released across the surface of the igniter plug from the central core to the outer case of the plug as a spark.



IGNITION SYSTEM COMPONENTS

Engine Start Modes

Location

The engine start modes are located within the EEC software.

Purpose

The purpose of the engine start modes is to provide a procedure to start the engine in all conditions. The EEC detects when water ingestion causes a significant effect on engine operation, and results in the selection of both igniters for the duration of the threat to maintain combustion.

Description

There are seven start modes as described below:

- Autostart.
- Manual Start.
- Automatic Inflight Start.
- Quick Relight.
- Autorelight.
- Dry Crank.
- Wet Crank.

Autostart

The EEC receives the start command from the operation of the cockpit controls discussed earlier. In this mode the EEC has full authority over the SCV, igniter selection and fuel input to the combustor. In Autostart the EEC detects and protects the engine against abnormal starts.

Manual Start

The EEC receives the start command from the operation of the cockpit controls and selection of manual start using the related manual start switch. In manual start mode the EEC detects and protects the engine against abnormal starts but the crew control SCV, fuel and igniter selections.

Automatic Inflight Start

Inflight, the Autostart sequence is part of the Autostart Mode except the EEC monitors for abnormal starts and the crew must carry out any corrective action if required.

Quick Relight

The quick relight function provides the facility to immediately relight the engine after inadvertent movement of the engine master switch from the ON to OFF and back to ON within a specified time.

Autorelight

The autorelight mode attempts to relight the engine should the EEC detect a flameout or if the engine goes below the normal idle conditions (sub-idle).

Dry Crank

The EEC receives the dry crank command from the combination and operating the cockpit controls discussed earlier. In this mode the EEC will operate the SCV and not allow fuel to the combustion chamber and will not operate the igniters.

Wet Crank

The EEC receives the wet crank command from the combination and operation of the cockpit controls discussed earlier. In this mode the EEC will operate the SCV and allow fuel to the combustion chamber but will not operate the igniters.

DRY CRANK

The EEC receives the dry crank command from the combination and operating the cockpit controls discussed earlier. In this mode the EEC will operate the SCV and not allow fuel to the combustion chamber and will not operate the igniters.

AUTO RELIGHT

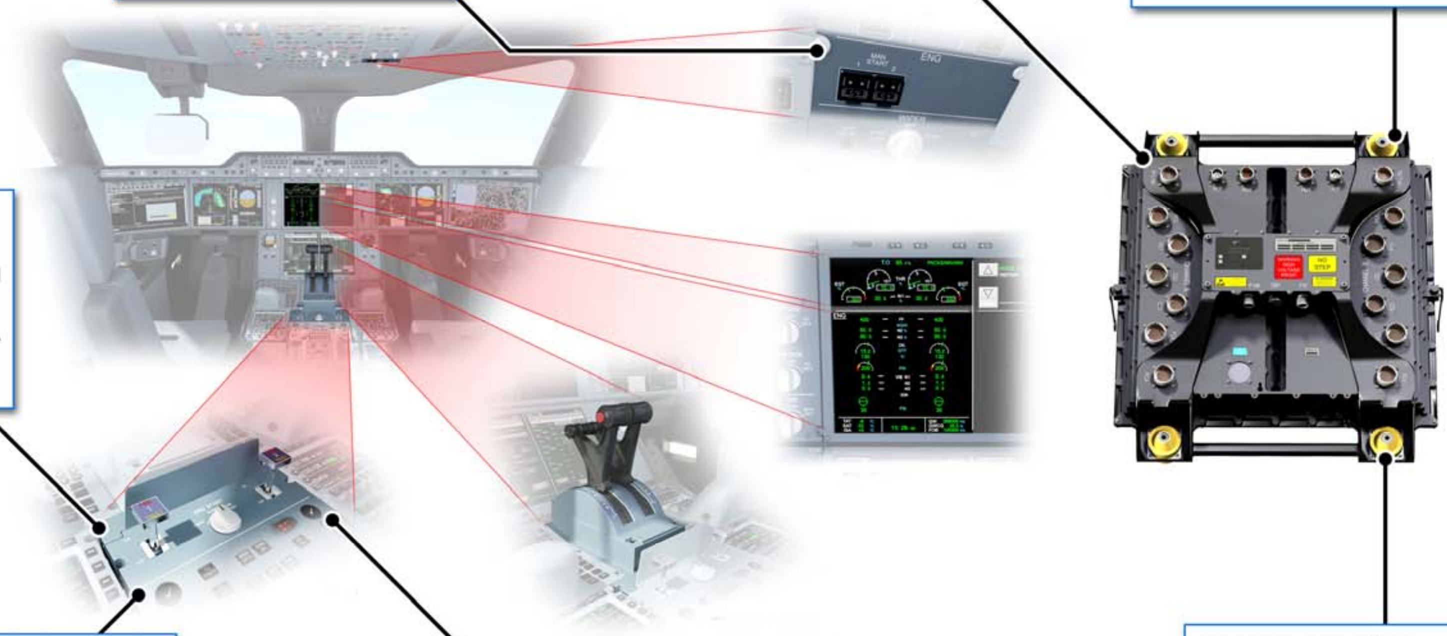
The autorelight mode attempts to relight the engine should the EEC detect a flameout or if the engine goes below the normal idle conditions (sub-idle).

AUTO START

The EEC receives the start command from the operation of the cockpit controls discussed earlier. In this mode the EEC has full authority over the SCV, igniter selection and fuel input to the combustor. In Autostart the EEC detects and protects the engine against abnormal starts.

WET CRANK

The EEC receives the wet crank command from the combination and operation of the cockpit controls discussed earlier. In this mode the EEC will operate the SCV and allow fuel to the combustion chamber but will not operate the igniters.



MANUAL START

The EEC receives the start command from the operation of the cockpit controls and selection of manual start using the related manual start switch. In manual start mode the EEC detects and protects the engine against abnormal starts but the crew control SCV, fuel and igniter selections.

QUICK RELIGHT

The quick relight function provides the facility to immediately relight the engine after inadvertent movement of the engine master switch from the ON to OFF and back to ON within a specified time.

AUTO INFLIGHT START

Inflight, the Autostart sequence is part of the Autostart Mode except the EEC monitors for abnormal starts and the crew must carry out any corrective action if required.

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ENGINE START MODES

Engine Start Operating Limits

General

The engine should only be operated within set limits. The engine should not be started or cranked outside these limits and if exceeded during operation, crews should take the necessary action to immediately return the operation to within the limits. All exceedances must be reported and recorded.

Exhaust Gas Temperature (EGT)

The engine during a ground start operation the EGT should be limited to 700 degrees centigrade when below 50% N3 speed.

During an in-flight start operation the EGT limit is increased to 900 degrees centigrade throughout the start sequence.

Oil and Fuel Temperature

The engine oil temperature should not be below -40 °c before starting or above 180 °c throughout the operating range.

The engine fuel temperature are limits are; minimum -54°c and maximum 55°c.

Tail Wind

An engine start should not be initiated if a tail wind is 10 Knots or greater.

Starter Motor Operation

Continuous operation of the starter motor must be limited in accordance with one of the following cycles:

1. Normal Cycle
    a. Up to 3 minutes continuous operation, then run down to zero N3 and allowed to cool for 30 seconds.
    b. Up to a further 3 minutes continuous operation then run down to zero N3 and allowed to cool for 30 seconds.
    c. Up to a further 1 minute continuous operation then run down to zero N3 and allowed to cool for 30 minutes.

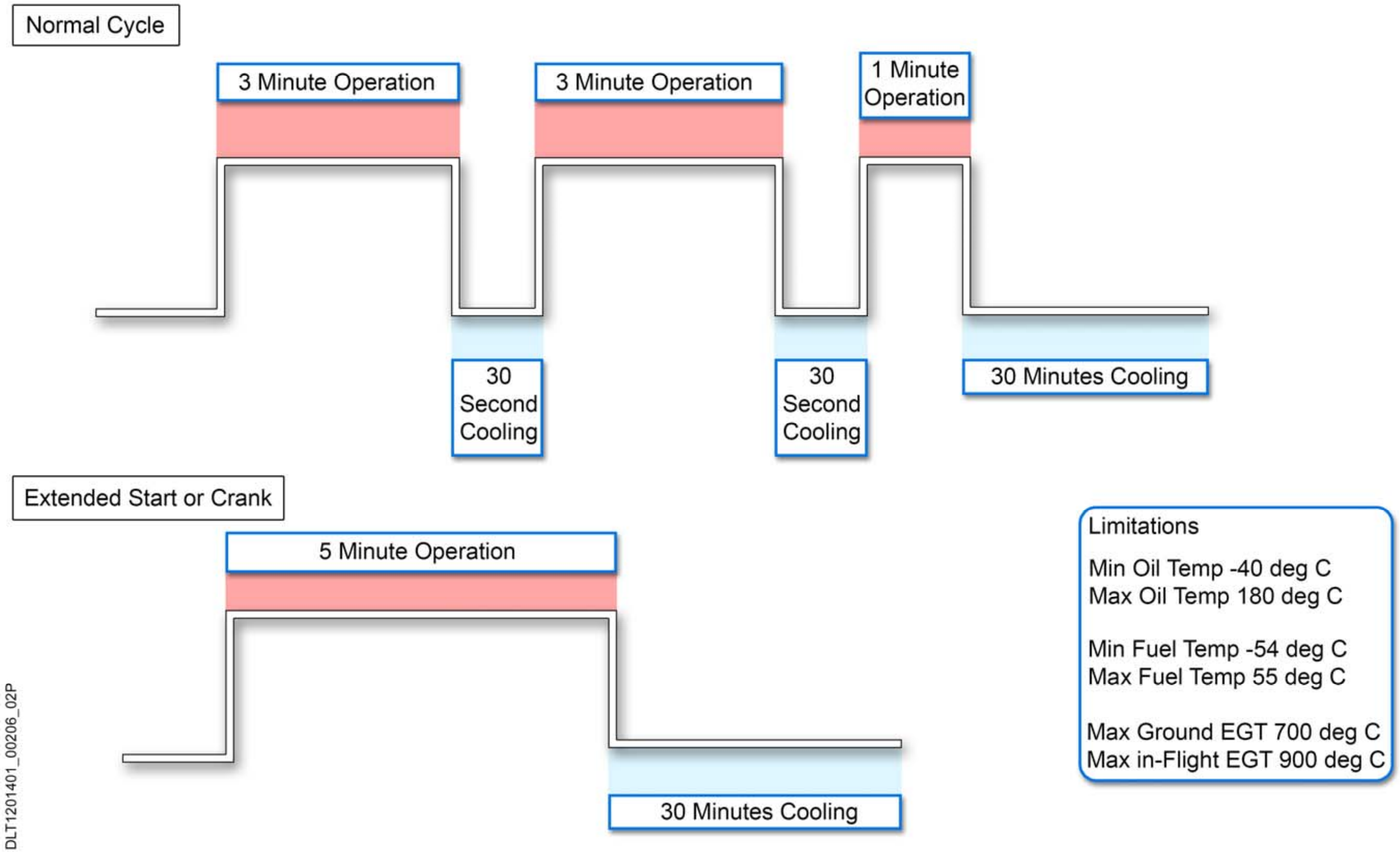
    Following two failed start attempts, further start attempts are not permitted until maintenance action has been carried out.

2. Extended Start Cycle

    Up to 5 minute continuous operation, then run down to zero N3 and allowed to cool for 30 minutes.

3. Extended Crank Cycle

    Up to 5 minute continuous operation, then run down to zero N3 and allowed to cool for 30 minutes.

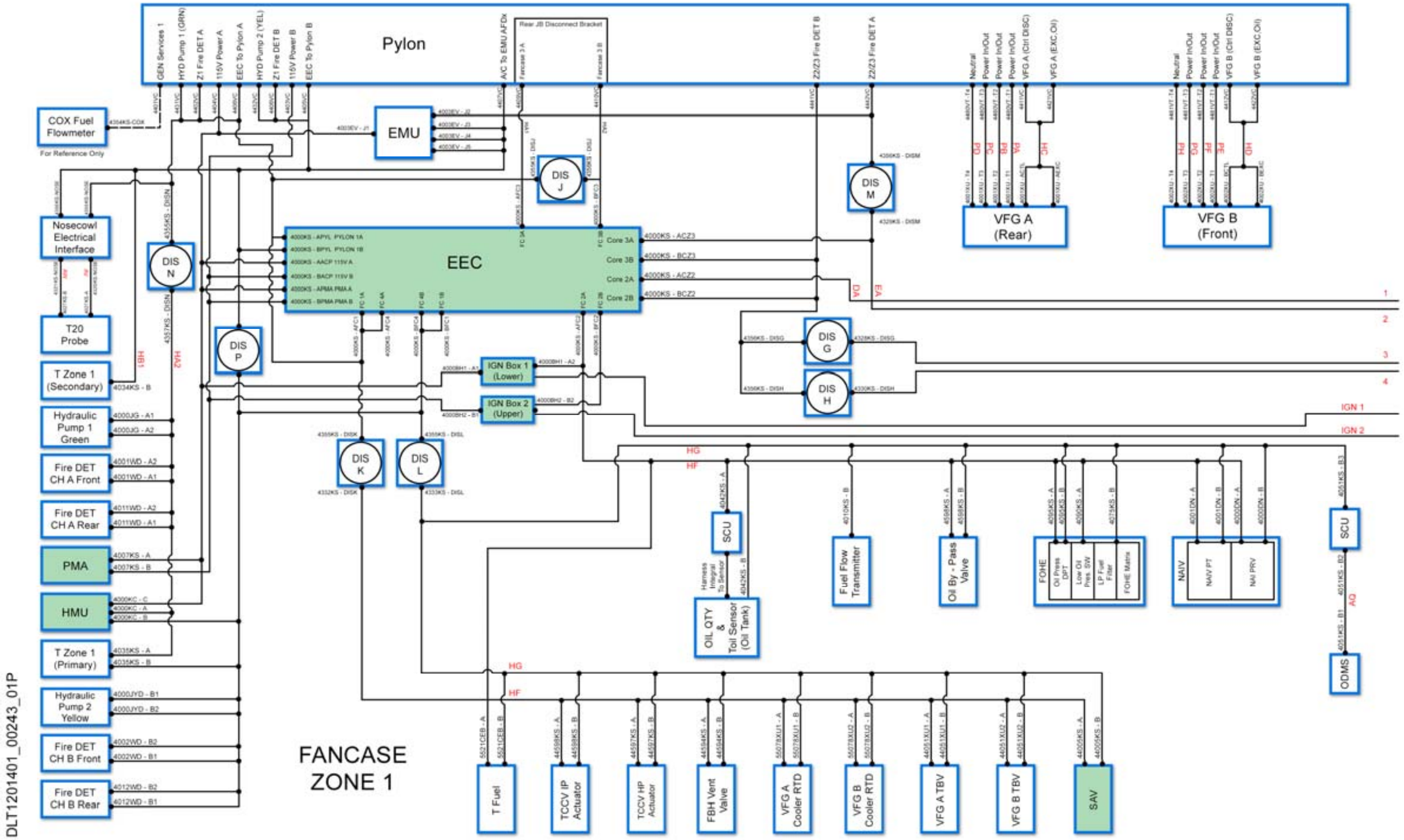


ENGINE START MODES

Starting and Ignition Systems - Wiring Diagrams Starting

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.

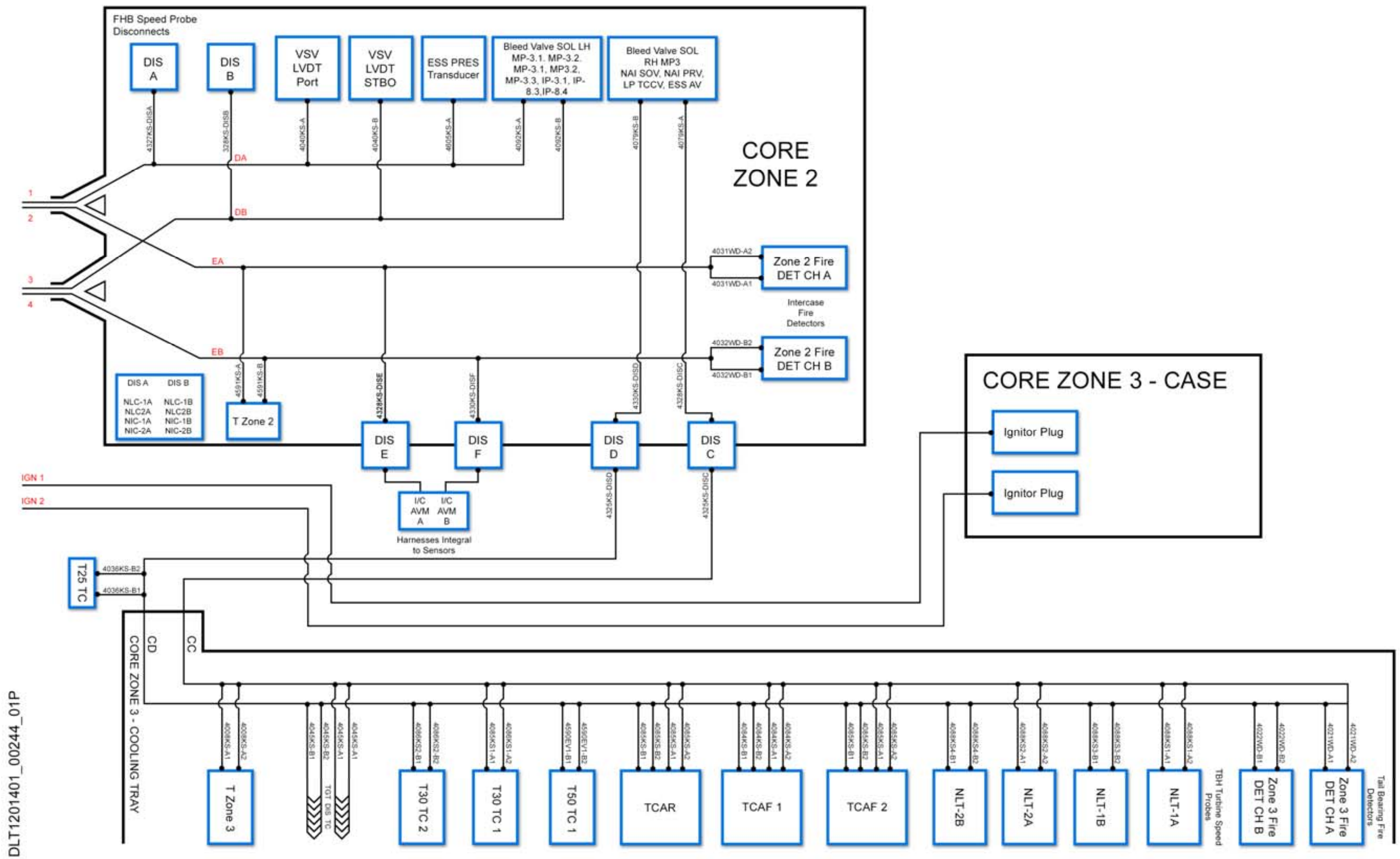


ZONE 1 WIRING DIAGRAM

Starting and Ignition Systems Wiring Diagrams

Introduction

Below is a wiring schematic of the associated units connected to the EEC. This is only a representation of the actual connectors but all plugs and connectors are indicated with the reference identifiers. This should be used for reference only and not be used as a definitive document. For a more detailed explanation (pins and sockets for troubleshooting purposes etc.) and for further details, then always refer to the actual wiring diagrams as given within the AMP or FIM electronic documentation.



ZONES 2 & 3 WIRING DIAGRAM

Section 13 – Engine Starting and Ignition System

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

- State the purpose of the engine Starting and Ignition Systems as fitted to the Trent XWB engine.
- Locate and identify the LRUs that form the engine Starting and Ignition Systems of the Trent XWB engine.
- Describe the purpose and operation of the LRUs that form the engine starting and ignition systems of the Trent XWB engine.
- State the WARNINGS and CAUTIONS associated with the engine starting and ignition systems of the rent XWB engine.
- Describe how the Trent XWB engine Starting and Ignition Systems interface with other engine and aircraft systems.

End of Engine Starting and Ignition System

Section 14 - Onboard Maintenance System (OMS) & Basic Troubleshooting

Section 14 – On-board Maintenance System (OMS) & Basic Troubleshooting

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

Identify the components, their purpose, location and interfaces of the On-board Maintenance System (OMS) & Basic Troubleshooting.

- Describe the purpose and operation On-board Maintenance System (OMS).
- Describe how the Trent XWB On-board Maintenance System (OMS) interfaces with A350 aircraft.
- Describe the unique fault code event identifier used on the A350 aircraft.
- Describe using the OMS the Basic Trouble Shooting Work Flow process of the A350 aircraft.

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On-board Maintenance System

Purpose

The On-board Maintenance System (OMS) is designed to support line & base maintenance activities and engineering follow-up by providing six features:

1. Failure reporting
2. System testing
3. Software uploading / downloading
4. Maintenance documentation access
5. Data monitoring of aircraft / engine systems
6. Managing aircraft configuration

Location

The OMS is part of the Onboard Information System (OIS) and is hosted in the Avionics Server Function Cabinet (ASFC), and consists of the following components:

- The Central Maintenance System Aircraft Control Domain (CMS-ACD)
- The Data Loading & Configuration System - Aircraft Control Domain (DLCS-ACD)
- Aircraft Condition Monitoring System (ACMS)
- Power Distribution Monitoring & Maintenance Function (PDMMF) (ATA 24 electrical systems and not covered by this course)

Description

The OMS is an electronic aircraft system, which deals with aircraft / engine systems maintenance and operational failure data. The OMS six features are grouped into three functions:

1. Diagnosis
2. Prognosis
3. Maintenance support

The diagnostic function which:

- Is designed to identify the root cause of a reported defect.
- Provides direct access to all maintenance data needed to correct reported defects.

The prognostic function which provides a means to:

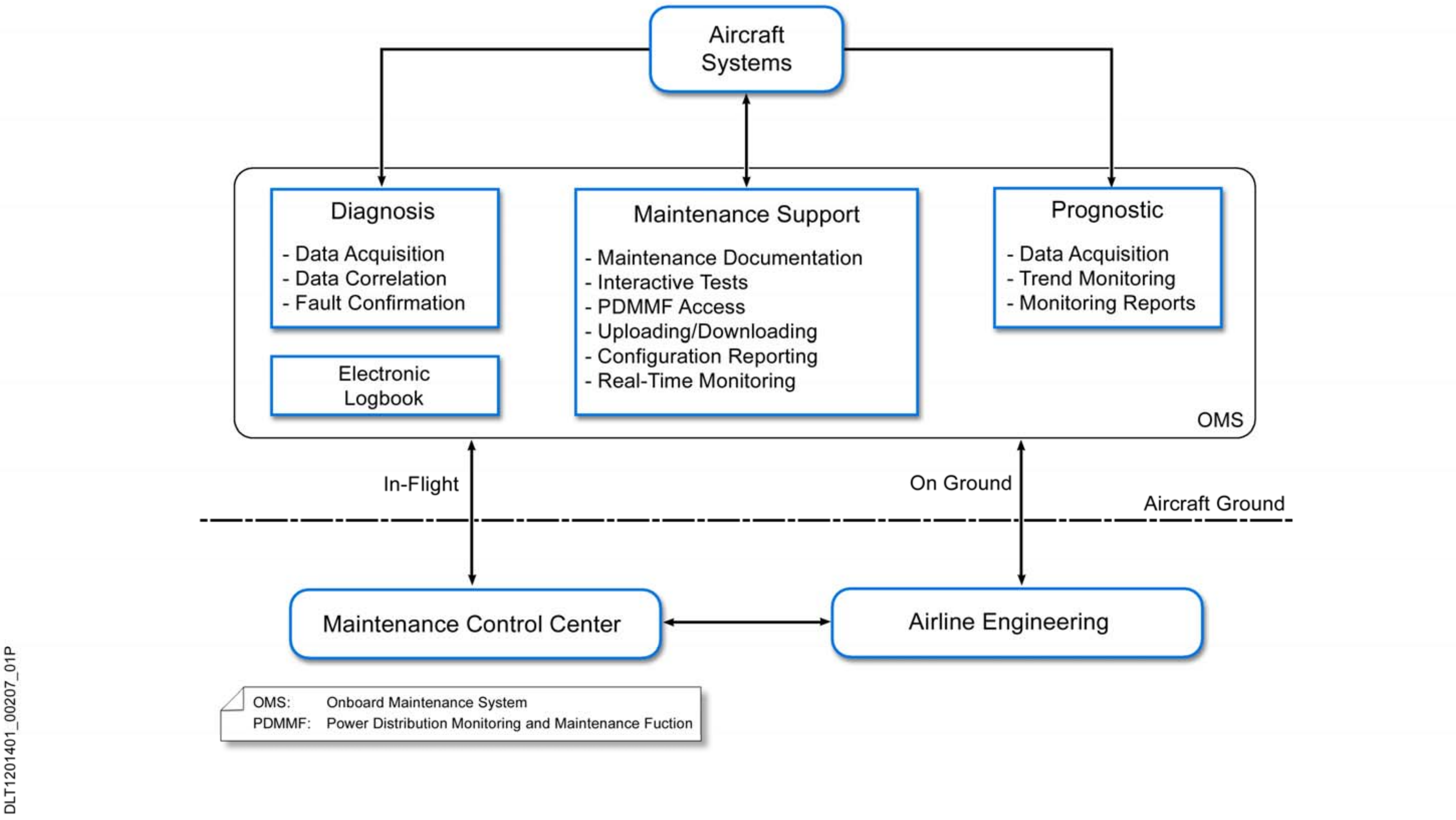
- Reduce scheduled maintenance
- Anticipate unscheduled maintenance

The support to maintenance activities In-flight and on-ground carries out the following:

- Centralizes data from the Built-In Test Equipment (BITE) of the various aircraft / engine systems
- Organizes data and creates standard or customized fault reports.

These reports are transmitted to the operational ground centers.

Ground maintenance personnel can consult and download the CMS reports and have direct access to the BITE of the various aircraft / engine systems.



ON-BOARD MAINTENANCE SYSTEM

OMS Interfacing

The OMS is linked to the aircraft / engine systems through two Secure Communication Interfaces (SCI) and to the Open world Server Function Cabinet (OSFC) through a Smart Diode Module (SDM) for human interfacing.

The OMS includes maintenance software applications that support aircraft maintenance tasks. The maintenance operators can get access to this data through Human-Machine Interfaces (HMI). The OMS can also send and receive data through various communication means.

Human Machine Interfaces (HMIs)

The maintenance applications are accessed and controlled through HMI's.

These are as follows:

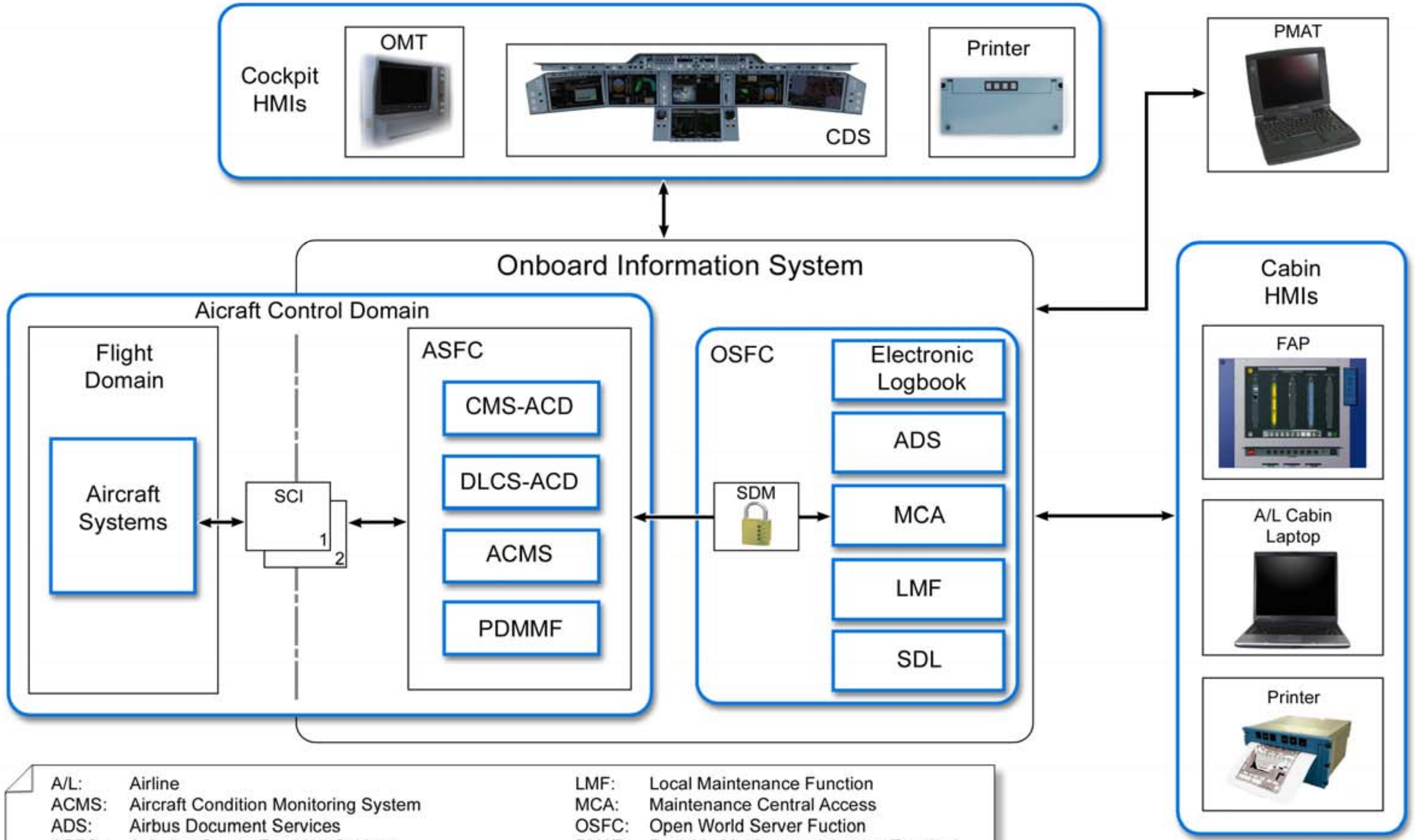
- Onboard Maintenance Terminal (OMT)
- CAPT & F/O outer displays of the Central Display System (CDS)
- Portable Multipurpose Access Terminals (PMAT)
- Flight Attendant Panels (FAP)
- Airline cabin laptop
- Printer

e-logbook and Maintenance Data Access and Recording Function

This optional function:

- Enables access to the Maintenance Data required to perform any maintenance task via the OMS.
- Enables the tracking of maintenance activities (e-logbook sub-function).
- Provides the aircraft technical status follow-up and lists all the maintenance actions that have been performed on the aircraft.

Integration of logbook within the OMS ensures that all maintenance related data is appropriately recorded and traced with reduced effort from the maintenance operator.



A/L:	Airline
ACMS:	Aircraft Condition Monitoring System
ADS:	Airbus Document Services
ASFC:	Avionics Server Function Cabinet
CMS:	Central Maintenance System
DLCS:	Data Loading and Configuration System
HMI:	Human Machine Interface
LMF:	Local Maintenance Function
MCA:	Maintenance Central Access
OSFC:	Open World Server Fuction
PMAT:	Portable Multipurpose Access Terminal
SDL:	Simple Data Loader
SDM:	Smart Diode Module
* =	Under Airline Responsibility

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OMS INTERFACING

Onboard Maintenance System (OMS)

& Basic Troubleshooting (Engine Specific)

Introduction

The A350 aircraft uses an Avionics Full Duplex Switched Ethernet (AFDX) communications network for transmitting digital data between systems on the aircraft including the Engine Interface Function (EIF) and Engine Control System. This is the primary digital communication bus used between the aircraft and Engine Control System.

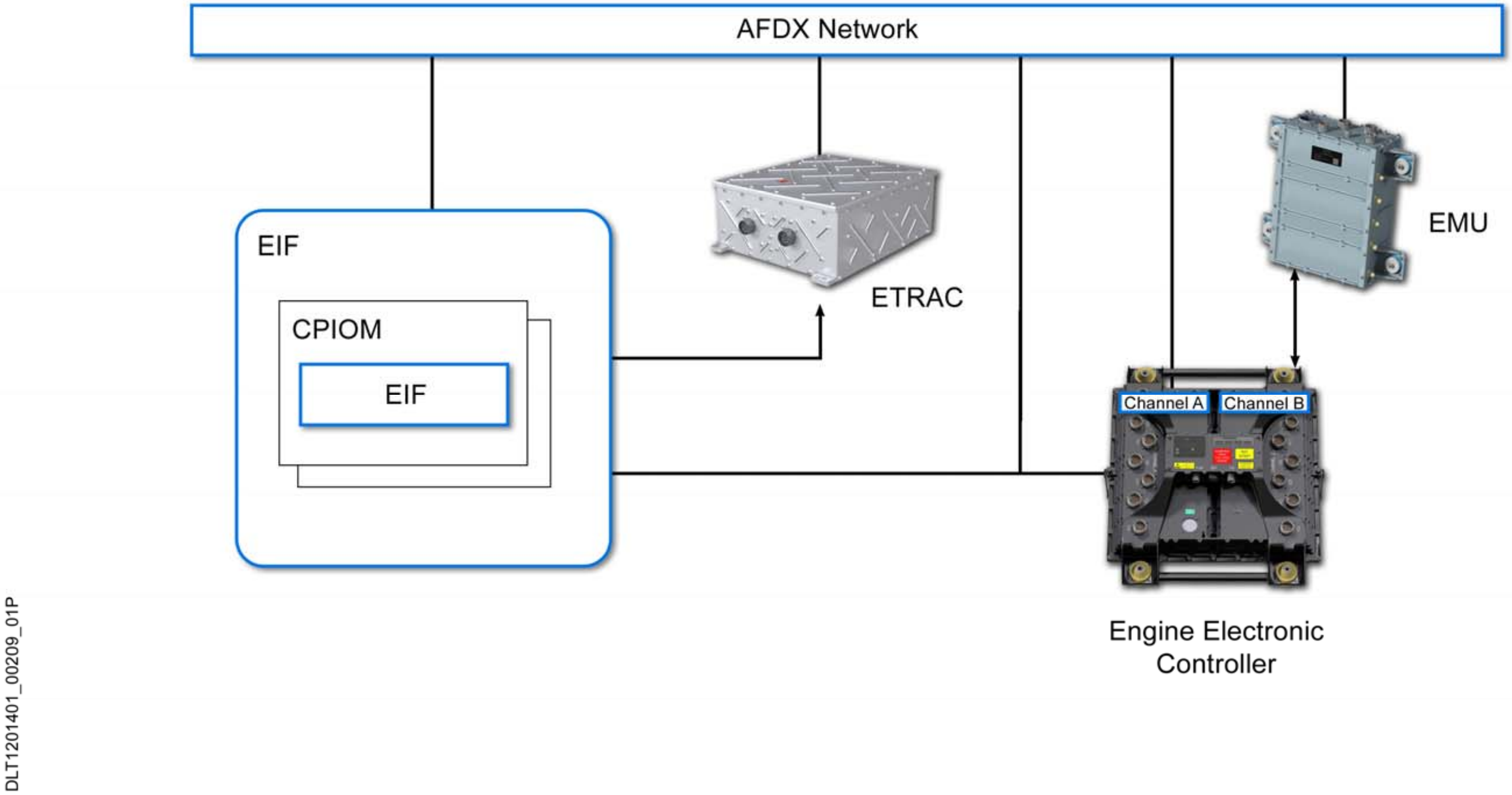
Each channel of the Engine Electronic Controller (EEC) has two bi-directional AFDX data buses for the communication of digital data between the EEC and the EIF.

The use of an AFDX communications network allows a high capacity of data to be transmitted between systems on the aircraft. The high payload capacity also allows improved fault detection capability of the data being transmitted on the network. An Application Layer Integrity Check (ALIC) is employed within the EEC software and the aircraft systems. This performs an additional cyclic redundancy check on the data transmitted between the engine and aircraft, and so increasing the overall integrity of the data checking. The EEC also performs an application layer sequence number check on the data received from the aircraft as a further level of fault detection capability.

The design of the EEC/EIF interface uses this improved fault detection capability to incorporate new methods of integrating engine and aircraft functionality to simplify the overall design. This has resulted in the following key differences from previous engine/aircraft applications:

- Where parameter validation and selection is performed in the engine and aircraft – the aircraft system will now perform all selections based on aircraft and engine data sources to enable high integrity selection to be performed and prevent selection of any erroneous parameter.
- How the EEC validates digital data from the aircraft – high integrity checking of aircraft data will detect any erroneous data transmitted from the aircraft.
- How the aircraft system (EIF) validates digital data from the Engine Control System – high integrity checking of engine data will detect any erroneous data transmitted from the engine.

These factors allow parameter selection for key functions like air data to be performed in the EIF and the selected parameter communicated and use within the Engine Control System using only the high integrity checking of the transmitted AFDX data.



NETWORK INTERFACES

Maintenance Support

The maintenance function is responsible for the reporting of individual faults with the Engine Control System and providing the necessary alerts and dispatch conditions related to the effect of faults on the operational status of the engine and Engine Control System.

The maintenance function interfaces with the aircraft to allow interactive maintenance operations to be performed with the engine and EEC whilst the engine is on the ground.

On-Board Maintenance

The EEC detects faults based on their effects on the engine, its systems and on aircraft operation. The EEC allocates detected faults to failure messages in preparation for onward reporting to the aircraft Centralized Maintenance System (CMS).

To isolate the failure in common fault symptoms, the EEC groups detected faults into the same failure message. The failure message allows the root cause of the problem to be addressed by maintenance personnel.

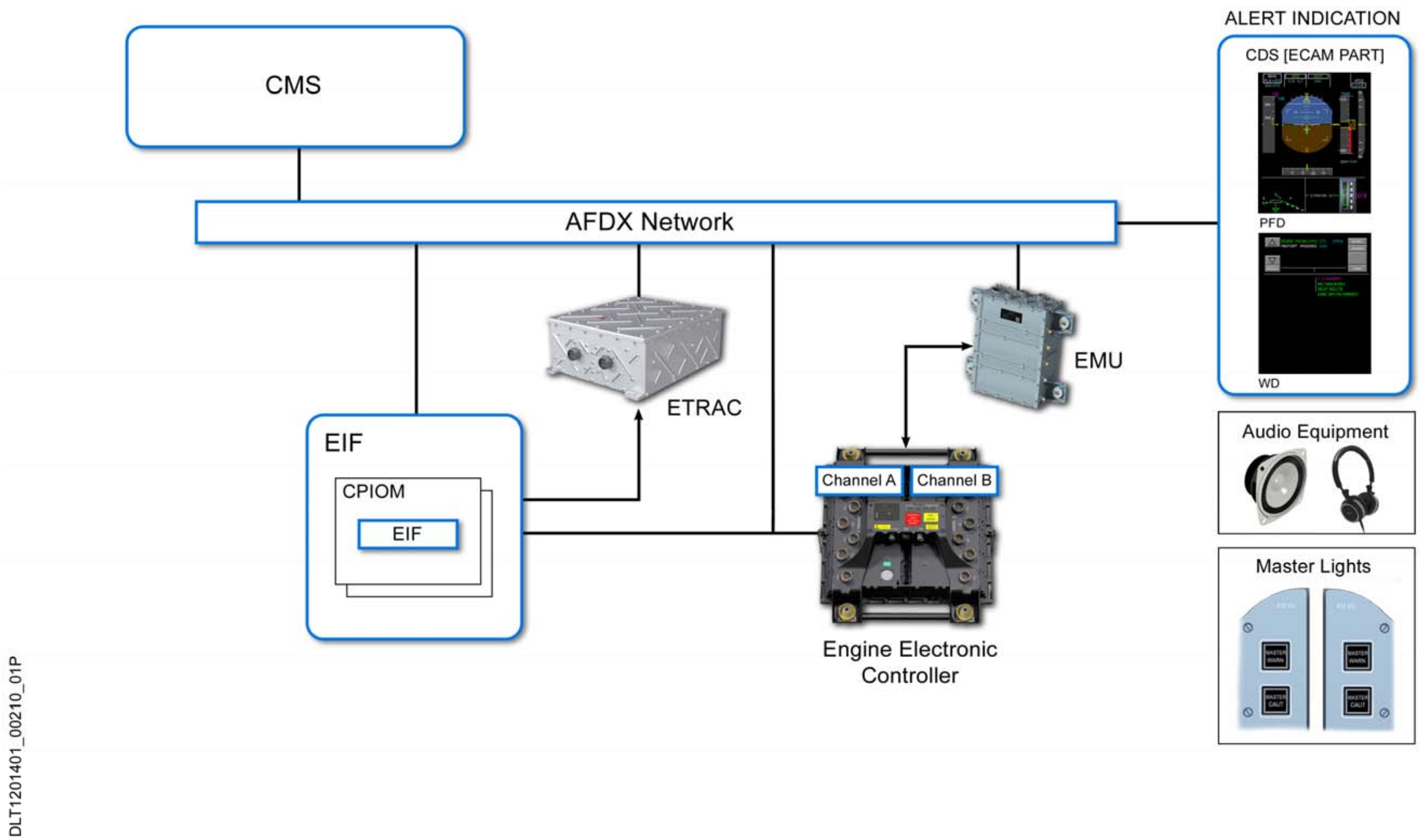
The EEC confirms failure messages in order to avoid nuisance detection and where required, the message is suppressed if the symptom is the result of another reported fault condition. Data associated with each failure message is collated to assist in determining the operating condition of the engine at onset of the failure.

Fault Reporting and Storage

The confirmed failure messages are processed into the format required by the Aircraft and transmitted to the CMS via the EECs AFDX outputs. The CMS records the failure messages and uses them to start maintenance corrective action. The EEC continuously reports the messages to the CMS until they clear. Each channel of the EEC reports the same failure messages.

Status Messages and Dispatch

The EEC first determines the effect of individual and a combination of failure messages, and then EEC allocates each combinational failure scenario to a particular dispatch category and generates an output to be transmitted to the aircraft Electronic Centralized Aircraft Monitoring (ECAM) via the EECs AFDX outputs. The ECAM system displays: flight deck effect engine parameters, their validity, alerts and associated procedures.



FAULT REPORTING

Interactive Maintenance

Interactive maintenance is provided by the aircraft and EEC interface to assist maintenance personnel to confirm, in a safe environment, the presence of faults and to verify that maintenance actions have returned the system to correct operation.

In addition, the EEC assists by semi-automating certain maintenance procedures and allows data held within the EEC to be reported and where necessary reset.

The EEC inhibits access to Interactive Maintenance if:

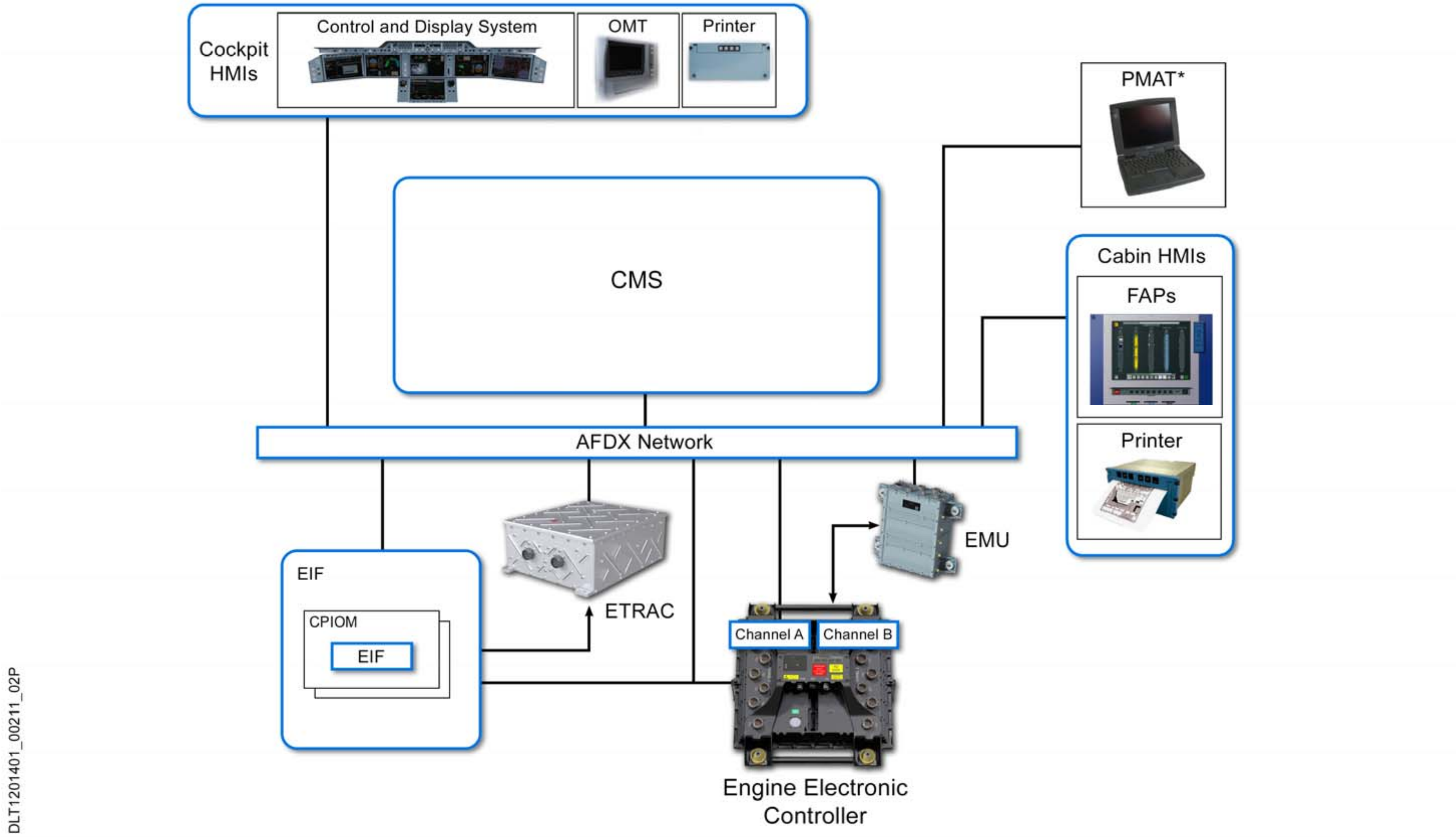
- The aircraft is not on the ground.
- The engine is running.
- The engine controls are not in a position corresponding to a static engine.
- One EEC channel is already in Interactive Maintenance.

Engine Monitoring Unit

The Engine Monitoring Unit is a digital electronic computer based unit mounted on the engine. The unit is a single channel device that receives analogue signals from the engine and EEC and has digital communication bus connections with the aircraft and EEC in order to:

- Provide vibration indications to the cockpit for continuous monitoring during engine operation.
- Provide data to allow fan trim balance maintenance operations.
- Support engine health monitoring.

The purpose of the Engine Health Monitoring System strategy is to provide predictive maintenance capability and to investigate un-planned events.



INTERACTIVE MAINTENANCE

Built-In Test Equipment (BITE)

Standard Description

BITE standard A & B

The A350 system BITEs integrate two different standards:

Standard A BITEs (e.g.: implemented on Satellite Communications (SATCOM), **V**ery High Frequency (VHF) **O**mnidirectional **R**ange (VOR), Distance Measuring Equipment (DME), Integrated Standby Instrument System (ISIS), etc.).

Standard B BITEs (e.g.: implemented on Flight Management Computer (FMC), Electronic Engine Control (EEC), Air Data / Inertial Reference Unit (ADIRU) etc.).

The Standard A BITEs are already used on AIRBUS Single Aisle (SA) and Long Range (LR) A/C.

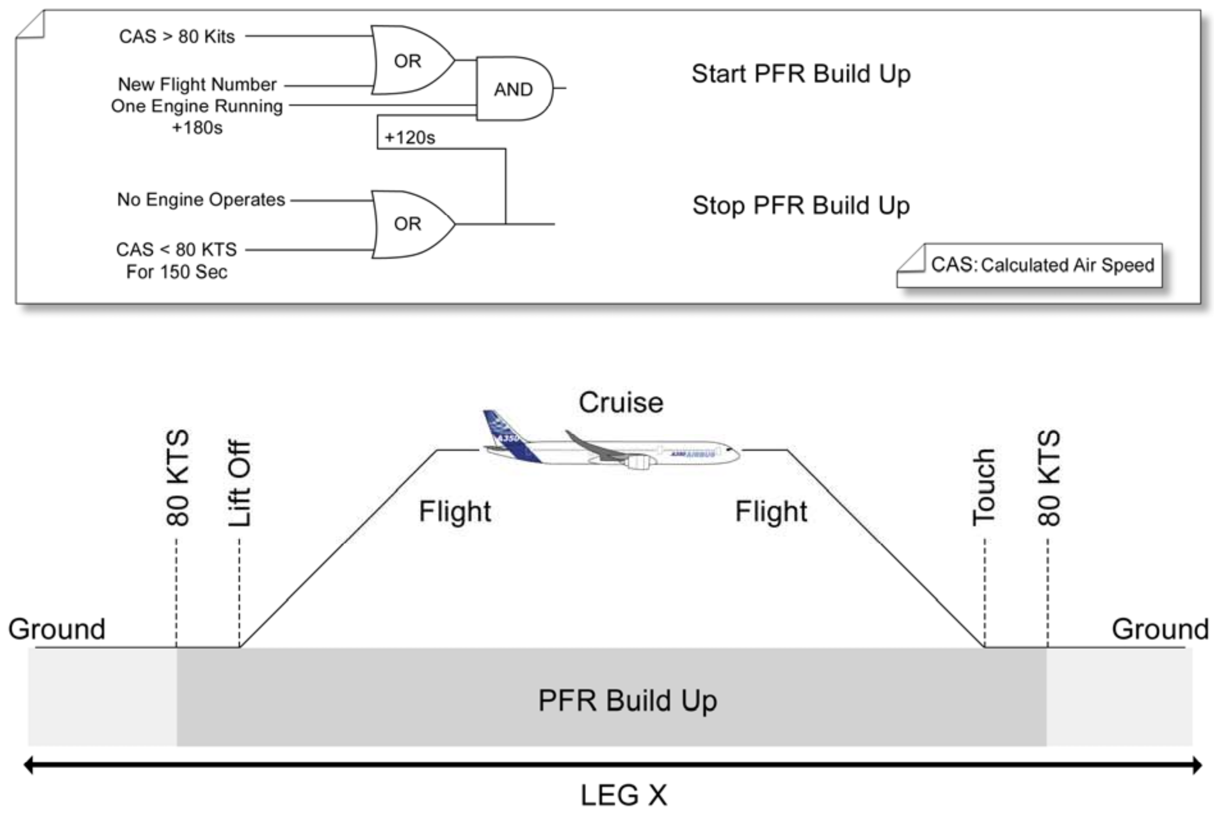
Note: The A350 computers, primary ATA 23 and ATA 34 system computers integrate Standard A BITEs.

The Standard A BITE has these specificities:

- The fault messages are transmitted in plain English to the CMS.
- Uses Aeronautical Radio, Incorporated (ARINC) 429.

The Standard B BITEs introduces these new features:

- The fault messages are transmitted to the CMS using fault codes.
- The system statuses are continuously monitored (refreshed status).
- Uses ARINC 429.



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BUILT-IN TEST

Built-In Test Equipment (BITE)

Fault code description

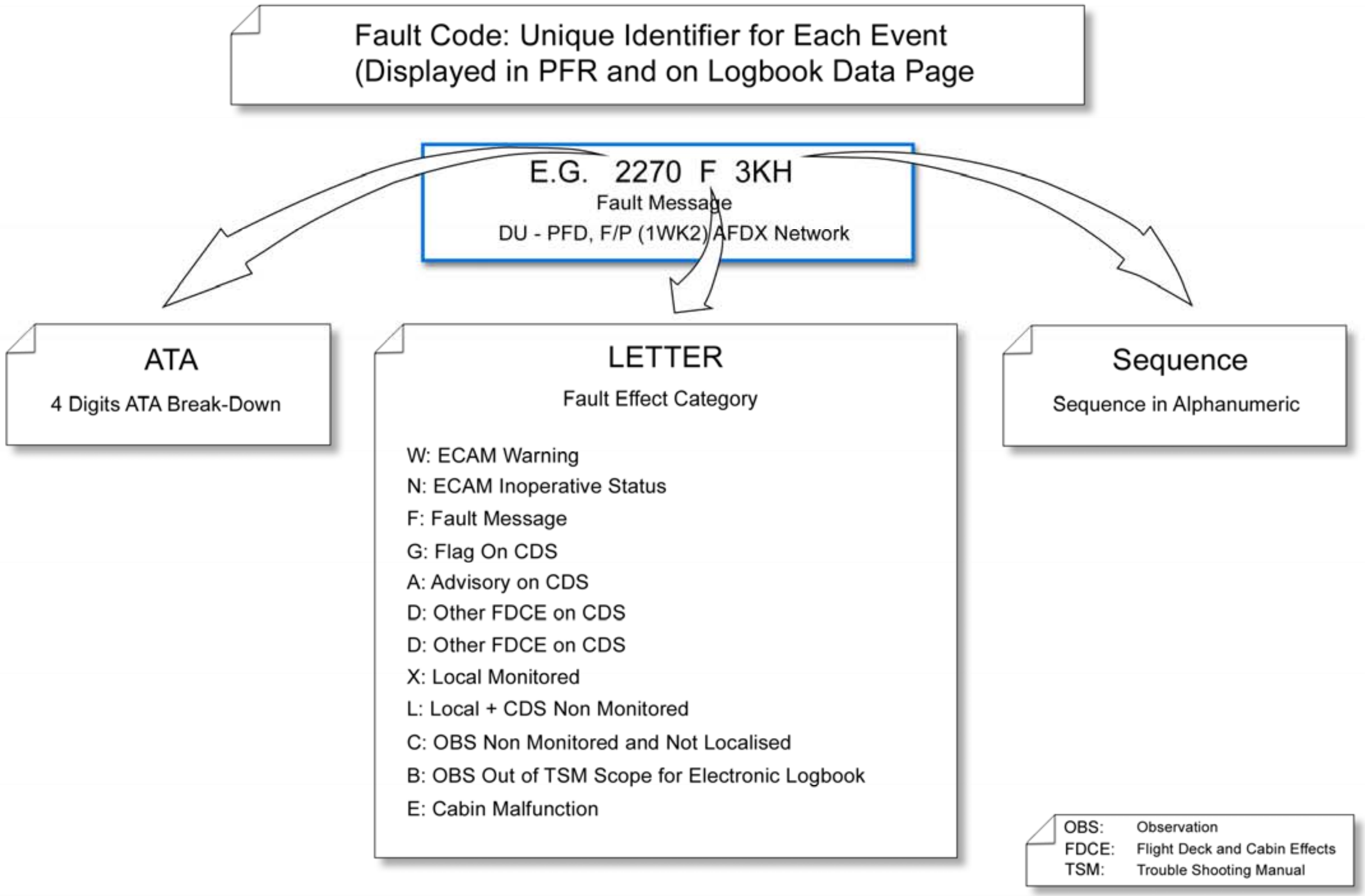
The fault code is the unique eight digit A/C event identifier.

It is composed of:

- Four first characters, which show the Air Transport Association (ATA) chapter related to the system.
- A middle letter, which shows the categorization of the event.
- Three last characters, which show the sequence of the event.

The Centralized Maintenance System (CMS) uses the fault codes for the correlation procedure between all received maintenance messages and cockpit effects to produce an associated fault case with related root cause and consequences.

The fault codes can be viewed in the fault-item detail page related to the Post Flight Report (PFR) on the OMS tool / application (entry point to get access to the PFR) and on the logbook entry page.



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OMS INTERFACING

Centralized Maintenance System (CMS) Data Recording During a Flight

During a flight, maintenance data received by the CMS are recorded in a time frame defined by several criteria.

The CMS data recording is launched if:

- If the last CMS data recording was stopped for a minimum of 120 sec and
- If a minimum of one engine operates for more than 180 sec and if:
    a. Calculated Air Speed (CAS) higher than 80 kts

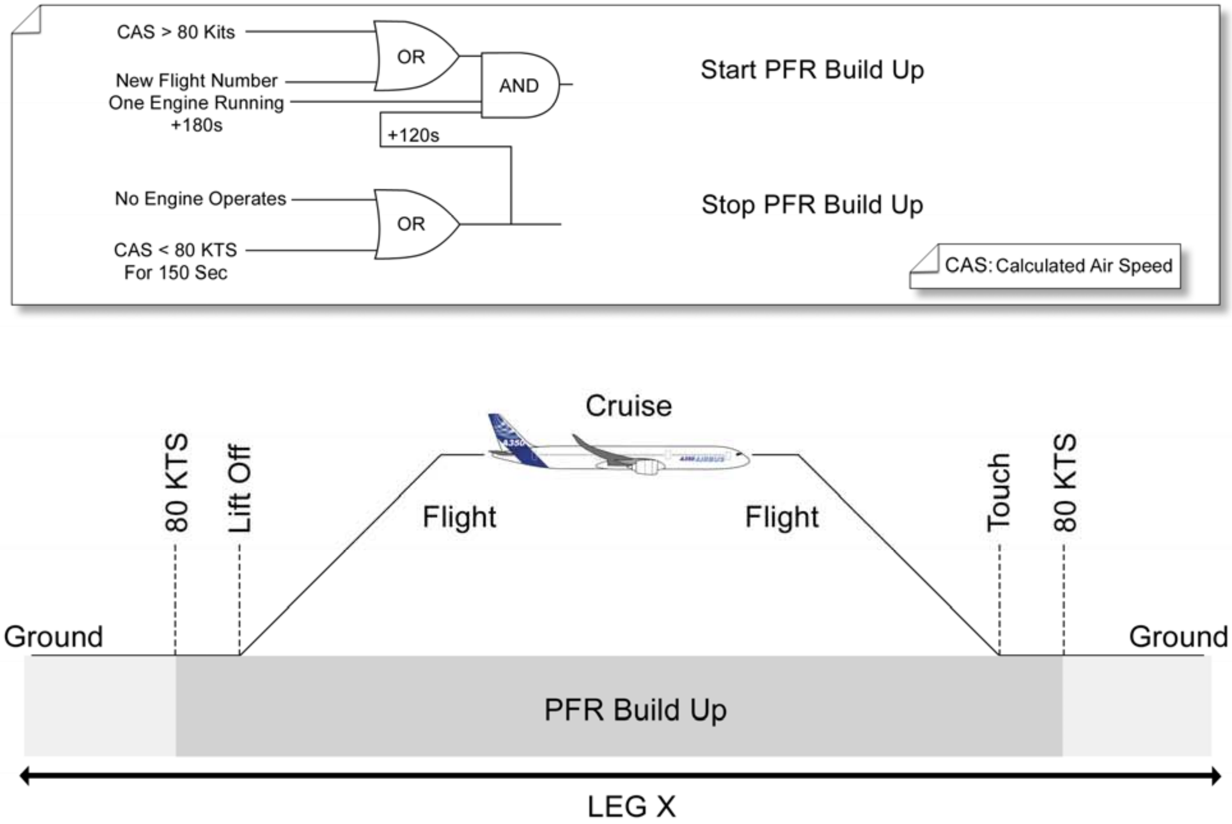
    or

    b. The pilots enter a new flight number.

The CMS data recording is stopped when:

- No engines are running
- CAS lower than 80 kts for 150 secs

So a failure that occurs outside the PFR recording phase will not be reported or shown on the PFR.



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OMS INTERFACING

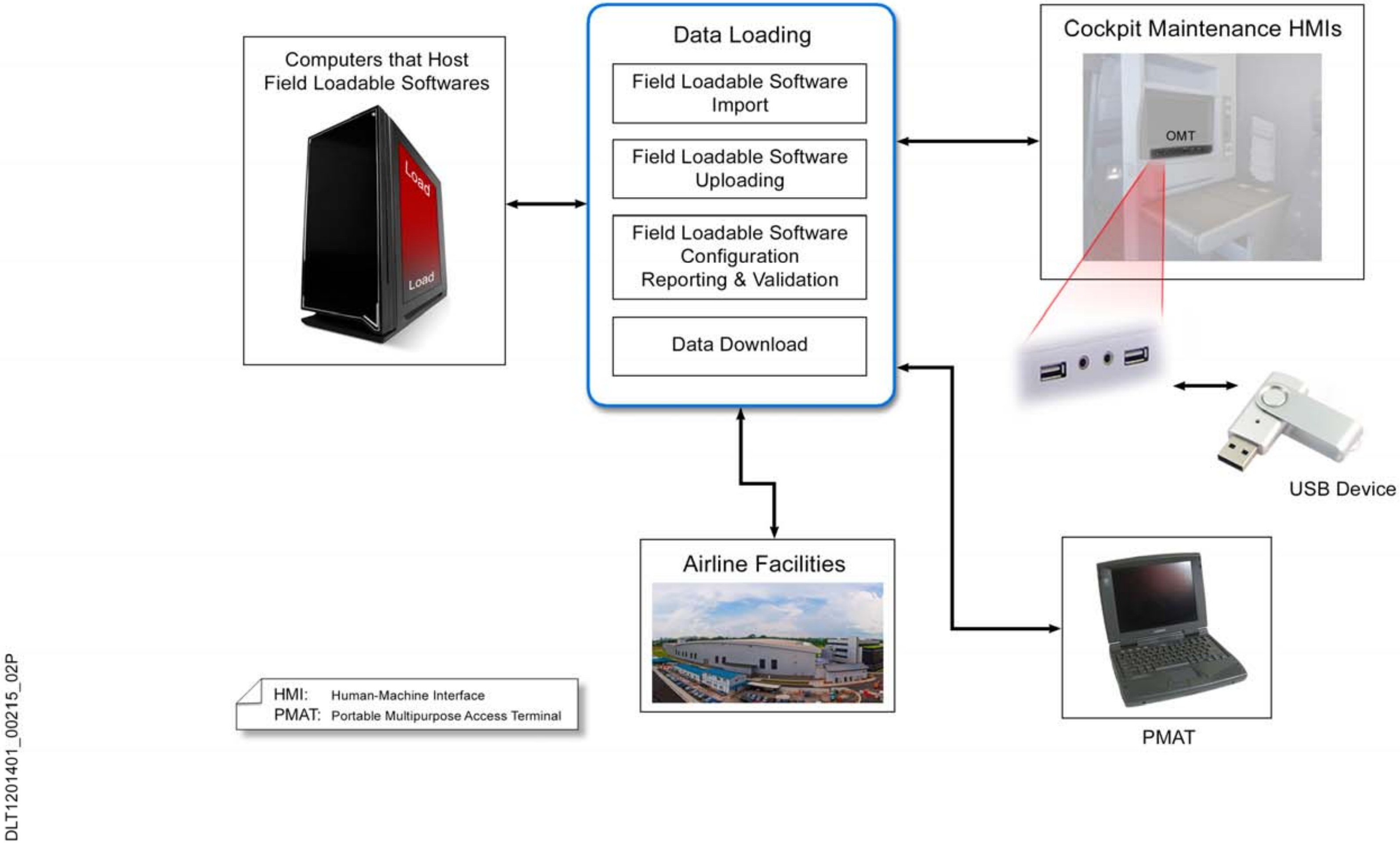
The Data Loading & Configuration System (DLCS)

Data loading Introduction

The data loading system operates the following functions:

- Field loadable software import from a USB device or from the ground
- Field loadable software uploading in the A/C computers
- Field loadable software configuration reporting and storage
- Field loadable software repository-management
- Data or file downloading to a USB device or to the ground.

All data loading functions are accessible through the Human Machine Interfaces (HMIs) of the Onboard Maintenance System (OMS), from the Onboard Maintenance Terminal (OMT), but also from the Onboard Information System (OIS) display and from the optional Portable Multipurpose Access Terminal (PMAT).



DATA LOADING & CONFIGURATION

Aircraft Condition Monitoring System (ACMS)

ACMS Introduction

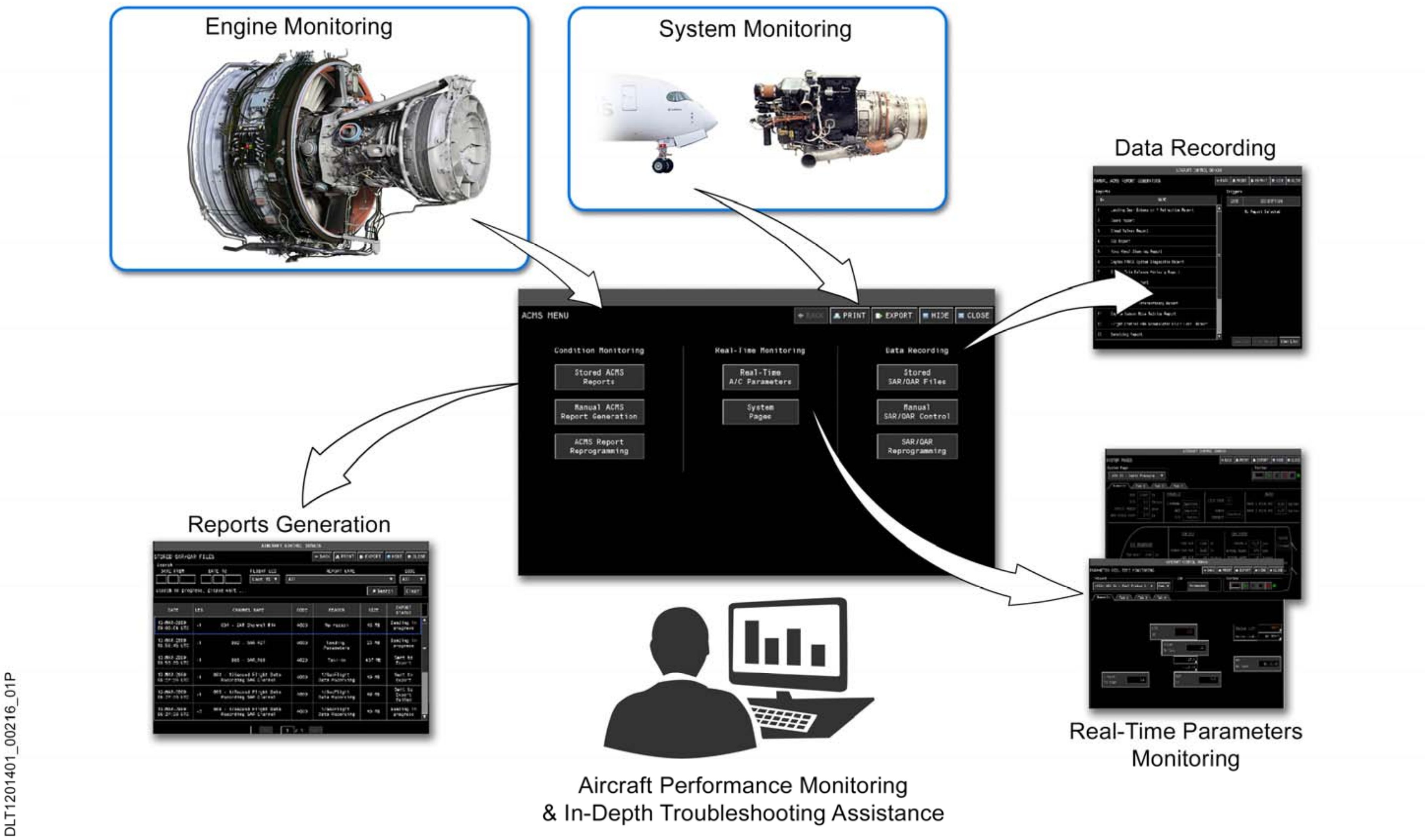
The primary function of the Aircraft Condition Monitoring System (ACMS) is to give scheduled, preventive maintenance and in-depth troubleshooting, through the monitoring of the efficiency and the degradation of the A/C systems and environment.

To achieve the above functions ACMS monitors:

- Engine condition
- A/C system condition
- A/C performance condition

To access these functions, the ACMS can:

- Generate reports (automatically or manually)
- Record data
- Display A/C parameters in real-time.



AIRCRAFT CONDITION MONITORING SYSTEM (ACMS)

Aircraft Condition Monitoring System (ACMS) Reports

ACMS reports groups

One of the functions of the ACMS is to give the A/C system reports for the system monitoring and in-depth analysis.

There are four groups of the ACMS reports:

- Airbus standard reports
- Airbus customer support reports
- Airline programmed reports
- On-board / on-ground reports.

The Airbus standard reports include:

- Engine reports (for trend monitoring and event analysis)
- System reports (for trend monitoring and event analysis)
- BITE related reports
- A servicing trend monitoring reports.

The Airbus customer support reports are dedicated to airline engineering, but only engineers of the Airbus Customer Services can change them through software uploading.

The Airline programmed reports can be changed by the Airline authorized staff through software uploading.

The on-board / on-ground reports can be changed by the Airline authorized staff through Aircraft Communication Addressing and Reporting System (ACARS) software uploading or through the on-board HMI.

The ACMS reports can be started through by the following methods:

- Automatic pre-specified triggering logic
- Manual request from on-board HMIs
- ACMS remote trigger-button
- Request from the ground Aircraft Communications Addressing and Reporting System (ACARS)
- Request from the CMS-ACD for the BITE correlation

The ACMS reports can be retrieved through:

- On-board Human Maintenance Interfaces (HMI) display consultation
- Transmission to the ground
- Downloading on the USB device
- On-board cockpit printer

ACMS Reports Groups

- A350 Standard Reports
- On-Request Airbus Reports
- Airline/Suppliers Programmed Reports
- Onboard/On-Ground Programmed Reports

ACMS Reports Triggering Conditions

- Automatic Predefined Trigger Condition
- Manual Request via HMI (OMT, PMAT, ETC.)
- ACMS Remote Trigger Button
- Request from the Ground (via ACARS)
- Request from CMS for Bite Correlation

ACMS Reports Data Retrieval

- Display on HMI (OMT, PMAT, ETC.)
- Transmission to the Ground
- Download on USB Key
- Via Onboard Cockpit Printer

ACARS: Aircraft Communication Addressing and Reporting System
HMI: Human-Machine Interface

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ACMS

Aircraft Condition Monitoring System (ACMS)

Smart Access Recorder / Quick Access Recorder Data

The ACMS records the A/C system parameters for visualization on the on-board HMIs or for recording.

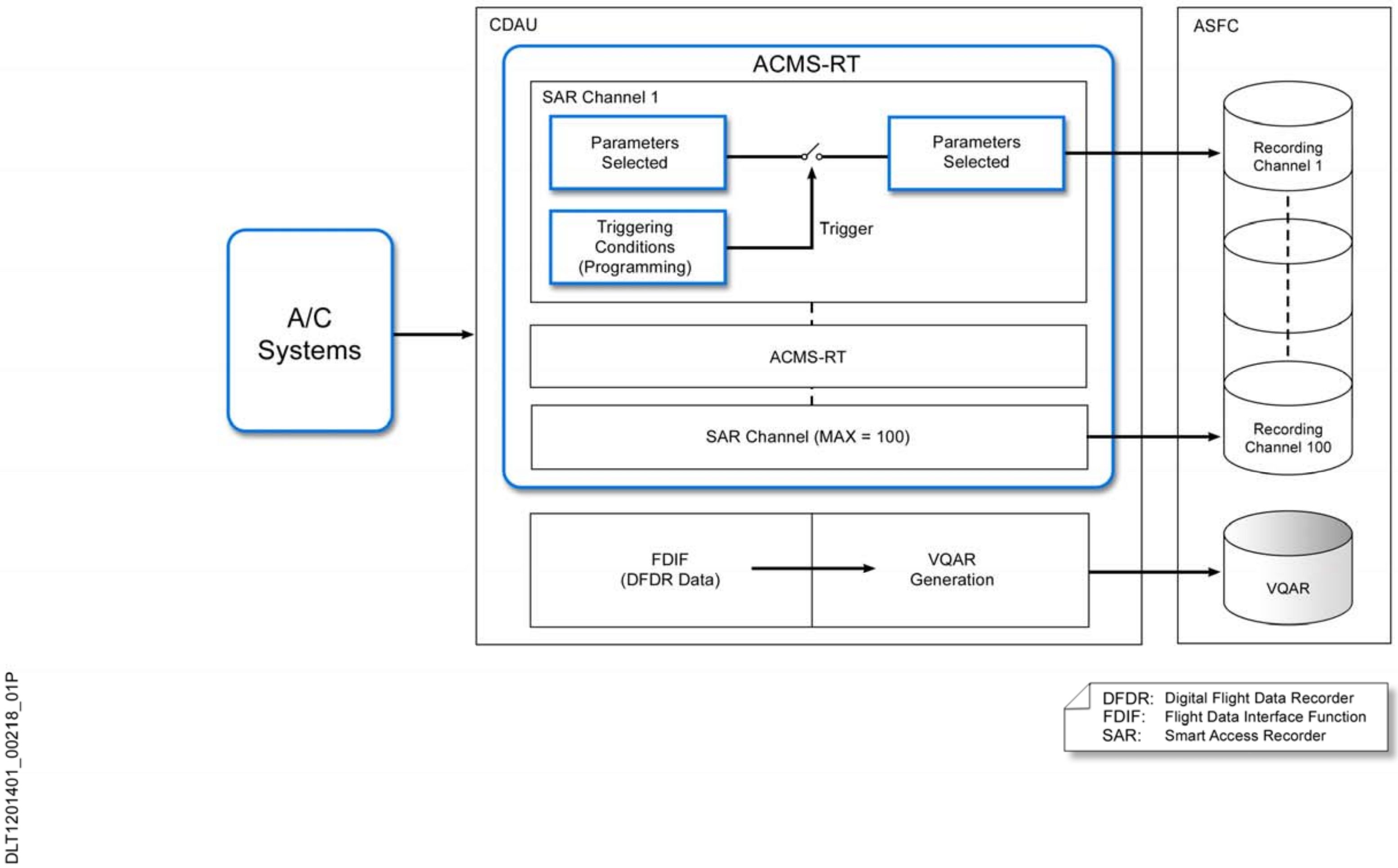
The recorded data is dedicated to the in-depth analysis for the troubleshooting procedures, or for prognostic activities at the airline engineering level.

Two types of data are recorded:

- Smart ACMS Access Recorder (SAR) data
- Quick Access Recorder (QAR) data.

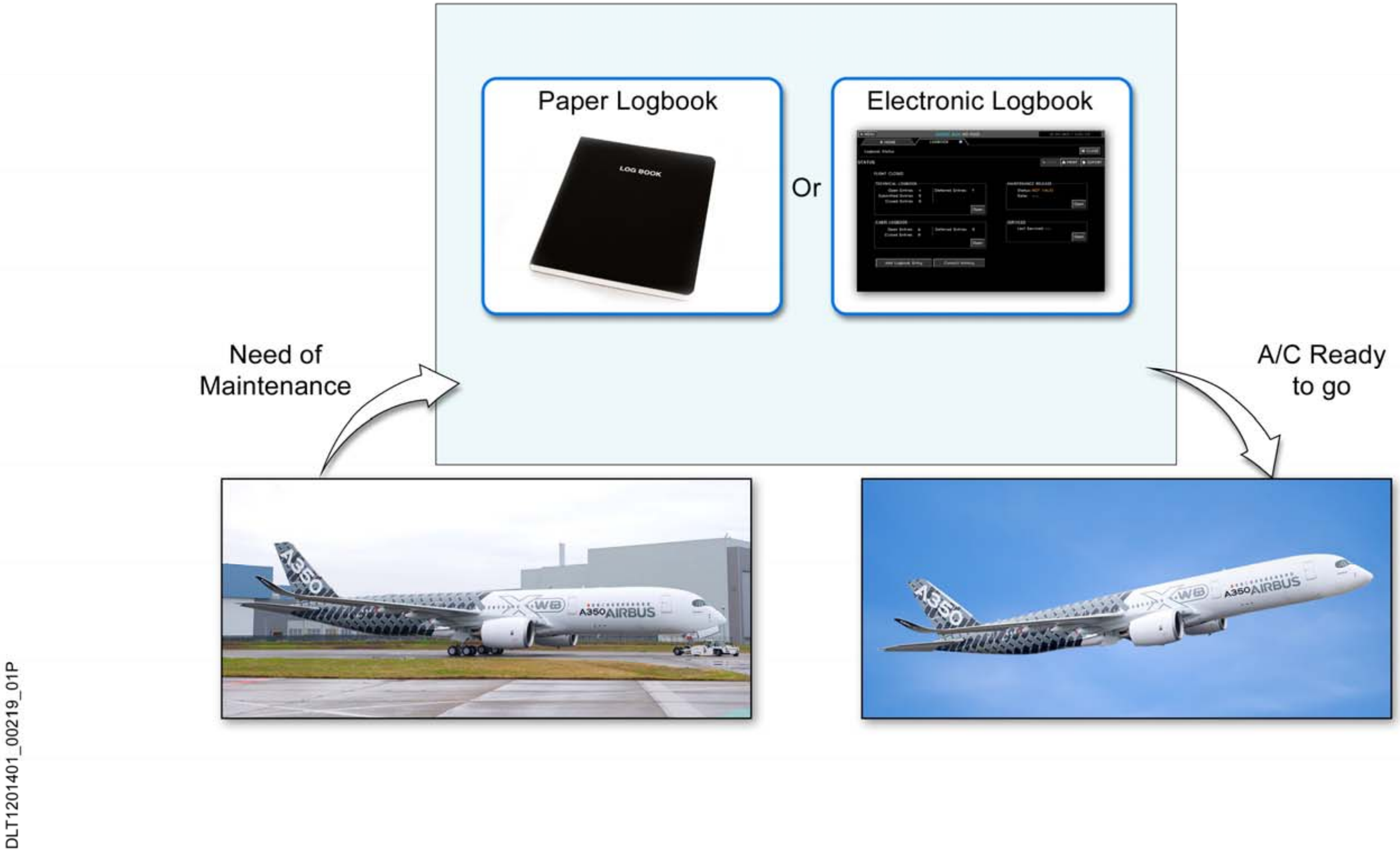
The SAR data is recorded in logic channels (100 channels max). Data is permanently received from A/C systems through the Centralized Data Acquisition Unit CDAU. Following the channel logic, dedicated parameters are selected, and if the triggering is achieved, the recording procedure is started. The data is compressed and sent to the ASFC (ASCM-SA) for recording and consultation through the on-board HMI.

The QAR data is received from the Flight Data Interface Function (FDIF) which does the collecting of the mandatory Digital Flight Data Recording System (DFDRS). A copy of this data is transmitted into the CDAU to a Virtual Quick Access Recorder (VQAR) recording function, which sends the data to the ASFC for recording in the VQAR database. The data can be downloaded or consulted from the on-board HMIs or can be sent to the ground.



AIRCRAFT CONDITION MONITORING SYSTEM (ACMS)

Basic Trouble Shooting Work Flow



BASIC TROUBLESHOOTING

Aircraft Logbook Consultation

After a flight, mechanics first check the A/C logbook to see if there are entries that come from the flight crew. The regulations impose that the mechanics must give an answer to each entry that the flight crews have made to release the A/C.

On the A350, two logbook configurations can be operated:

- Standard paper logbook
- Electronic logbook.

Paper Logbook Configuration

If the Electronic logbook configuration is not available, the paper logbook is used for the consultation of flight crew entries. To start the maintenance phase the following procedure should be used:

Open entry message:

If the technical logbook shows that there is an open entry message recorded in-flight. The mechanic must consult the Minimum Equipment List (MEL) to do the A/C dispatch assessment.

Dispatch message & Warning message:

Dispatch messages are used as an entry point into the MEL. Consult the dispatch page on the ECAM system display to find the dispatch message related to the logbook entry. A related warning message (if there is one) is shown on the ECAM warning display.

An example, of a dispatch message is (Integrated Modular Avionics (IMA) MULTIPLE RESOURCES); with a related warning message (ENG 2 REVERSER FAULT).

MEL & Dispatch condition status:

Access to the MEL to determine the dispatch condition status related to the flight crew entry.

There are two different ways to get access to the MEL:

a. Free access from the OMS documentation menu.
b. Direct access to the MEL correlated entry from a dedicated hyper link on the OMS Post Flight Report (PFR) page.

a. Free access to MEL:

Free MEL access from the OMS documentation menu.

First select dispatch message fault topic on the MEL home page. Then retrieve from the list shown on the ECAM the related dispatch message. Select the dispatch message to display its dispatch condition.

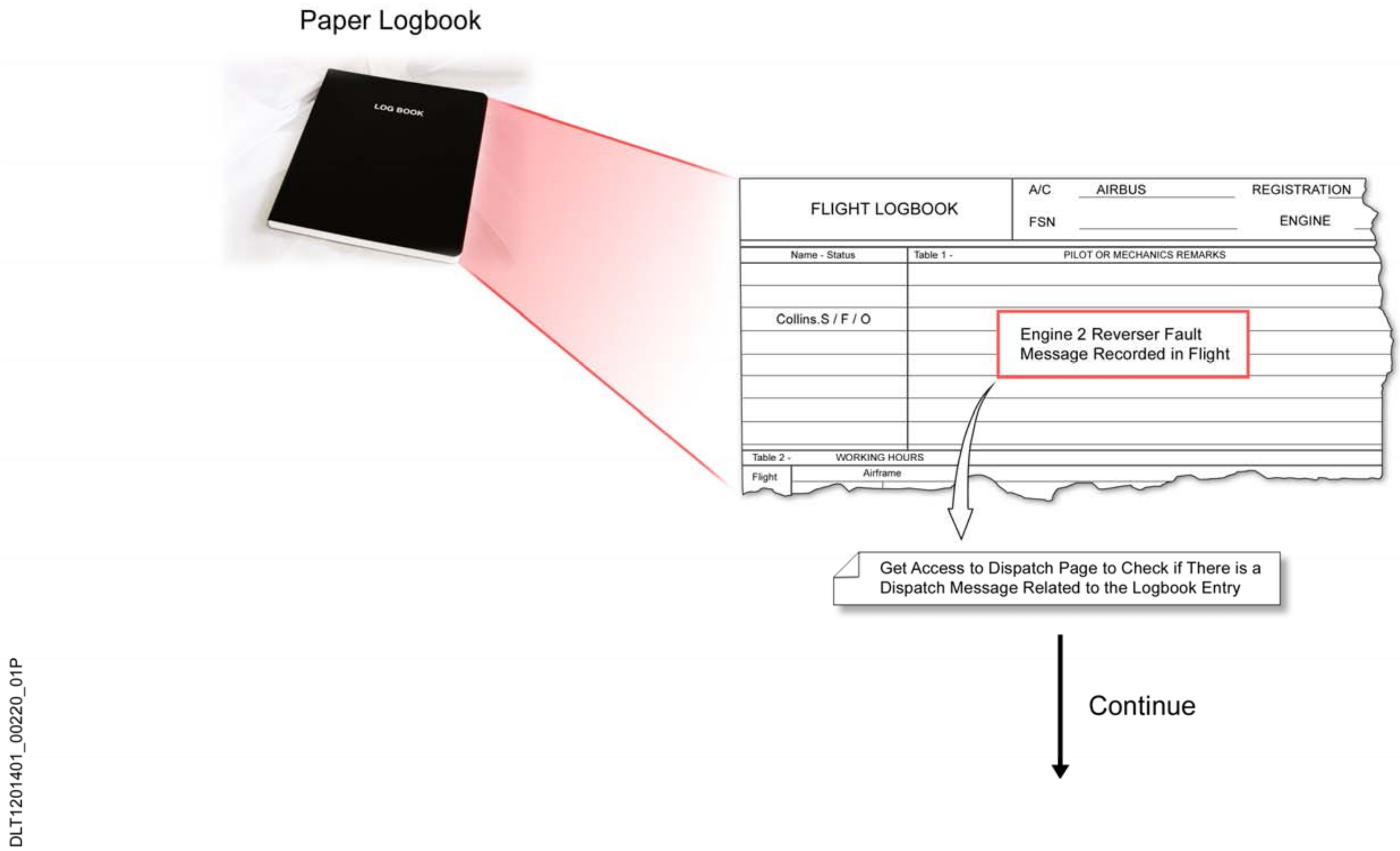
b. Access MEL from PFR:

Select the PFR from the OMS home page. Then select the warning / dispatch message related to the logbook entry (The effect detail page comes into view), on this page there is detailed information correlated to the fault, especially the correlated MEL entry. Select the hyperlink to get access to display its dispatch condition.

Dispatch assessment:

From the MEL correlated page, the mechanics can do the dispatch assessment related to the selected open entry.

If NO DISPATCH is indicated for the IMA - MULTIPLE RESOURCES dispatch message, the mechanic must repair the failure to make the maintenance release of the A/C possible.



BASIC TROUBLESHOOTING

From Last

Free Access to MEL from OMS Menu

OR

Access to MEL from OMS PFR Correlated Hyperlink

Continue

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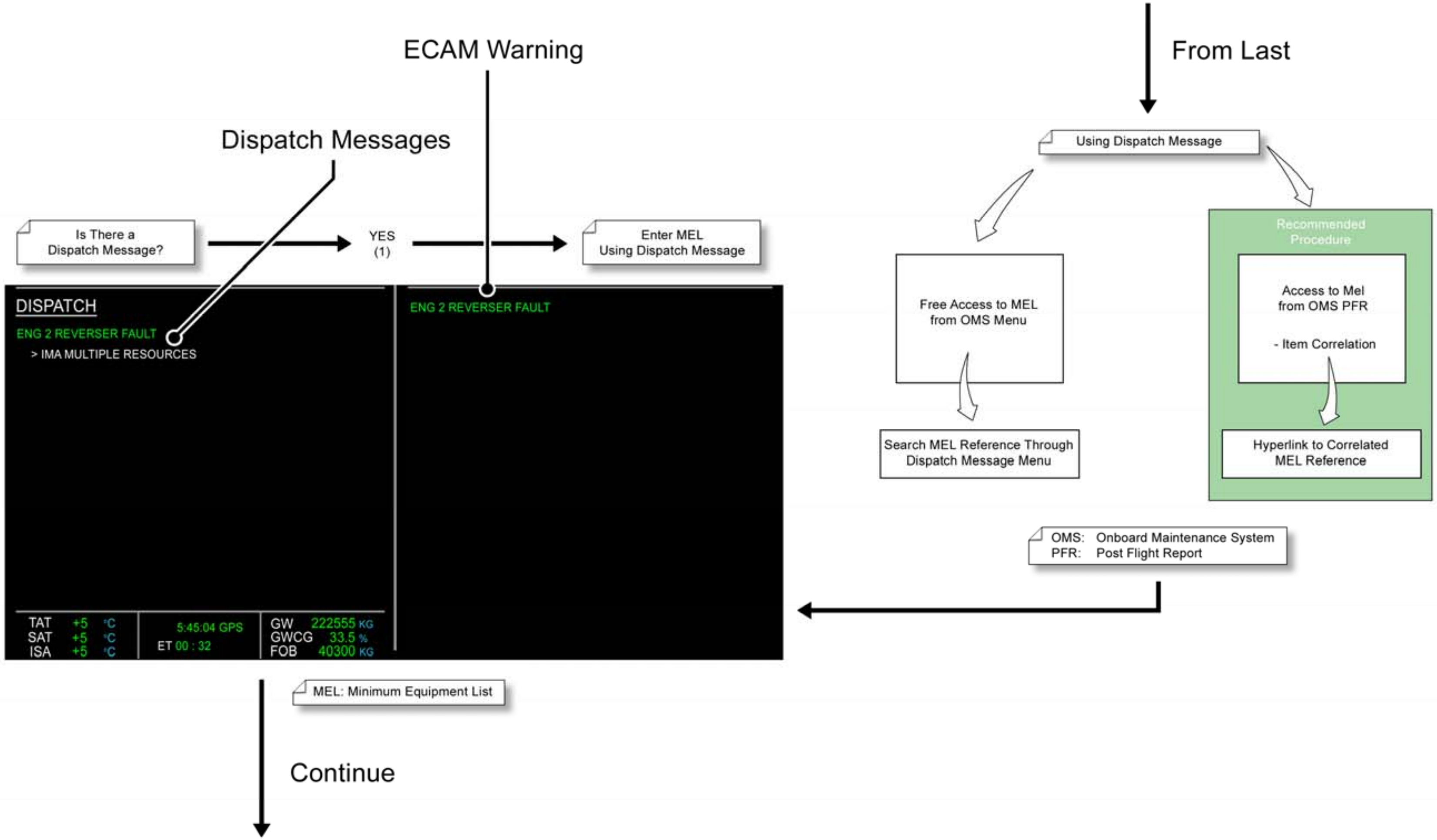
BASIC TROUBLESHOOTING



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BASIC TROUBLESHOOTING

From Last

Consult PFR on OMS & Start Troubleshooting

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BASIC TROUBLESHOOTING

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Electronic Logbook Configuration

If the electronic logbook is installed it can be used for the consultation of possible flight crew entries. To start the maintenance phase the following procedure should be used:

Electronic logbook status:

The mechanics can get access to the electronic logbook through the On-board Maintenance System (OMS), from the On-board Maintenance Terminal (OMT) or through the outer display units of the Control and Display System (CDS).

When the electronic logbook is used, it gives access to hyperlinks with the Minimum Equipment List (MEL).

If there is an open entry, the mechanics must consult the related entry before they start the dispatch assessment.

Note: since there is a minimum of one open entry; the maintenance release status is NOT VALID.

MEL access through hyperlink:

The technical report page gives access to the list of possible open and deferred entries that the flight crew made.

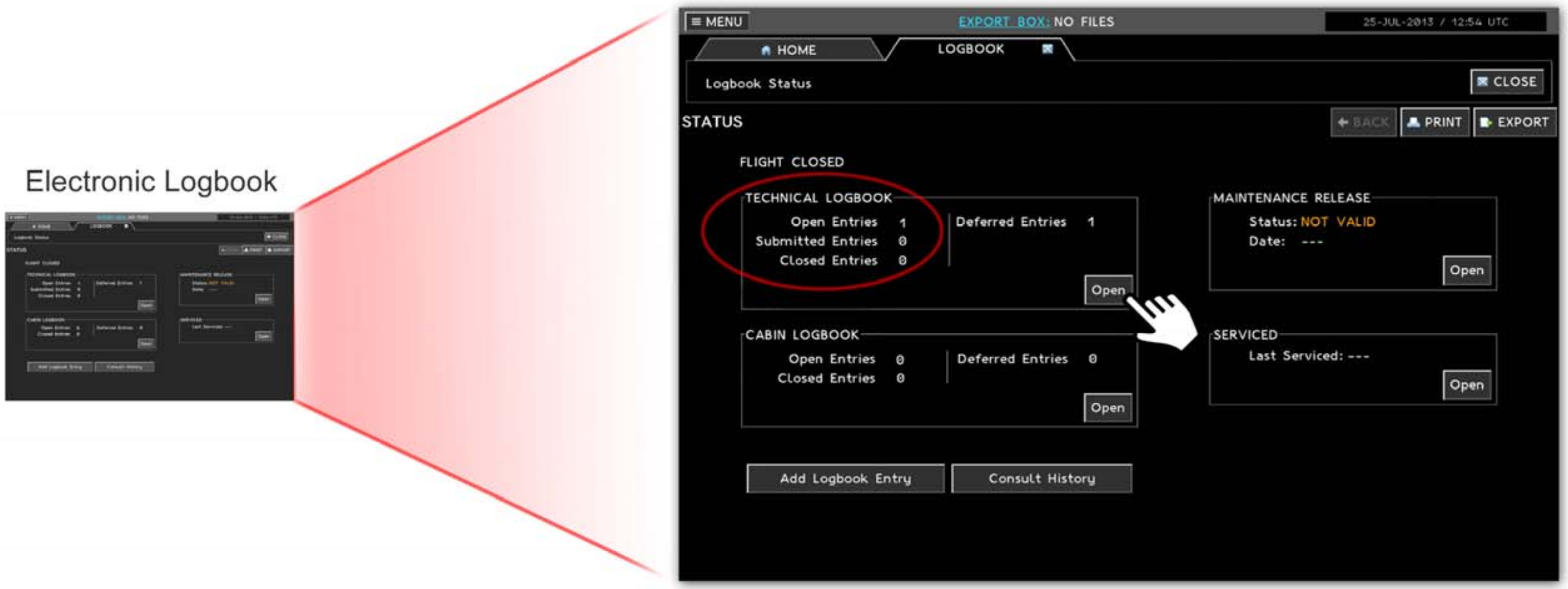
An example, of an open entry is:

(ENG 2 REVERSER FAULT > IMA MULTIPLE RESOURCES)

ENG 2 REVERSER FAULT is the ECAM warning related to the failure. IMA MULTIPLE RESOURCES is the dispatch message related to the failure. On this page, there is a hyperlink that gives direct access to the MEL/Configuration Deviation List (CDL) reference correlated with the selected open entry.

Dispatch assessment:

From the MEL correlated page, the mechanics can complete the dispatch assessment related to the selected open entry. So if the related condition of dispatch is NO DISPATCH, the mechanics must repair the failure to make the maintenance release of the aircraft possible.



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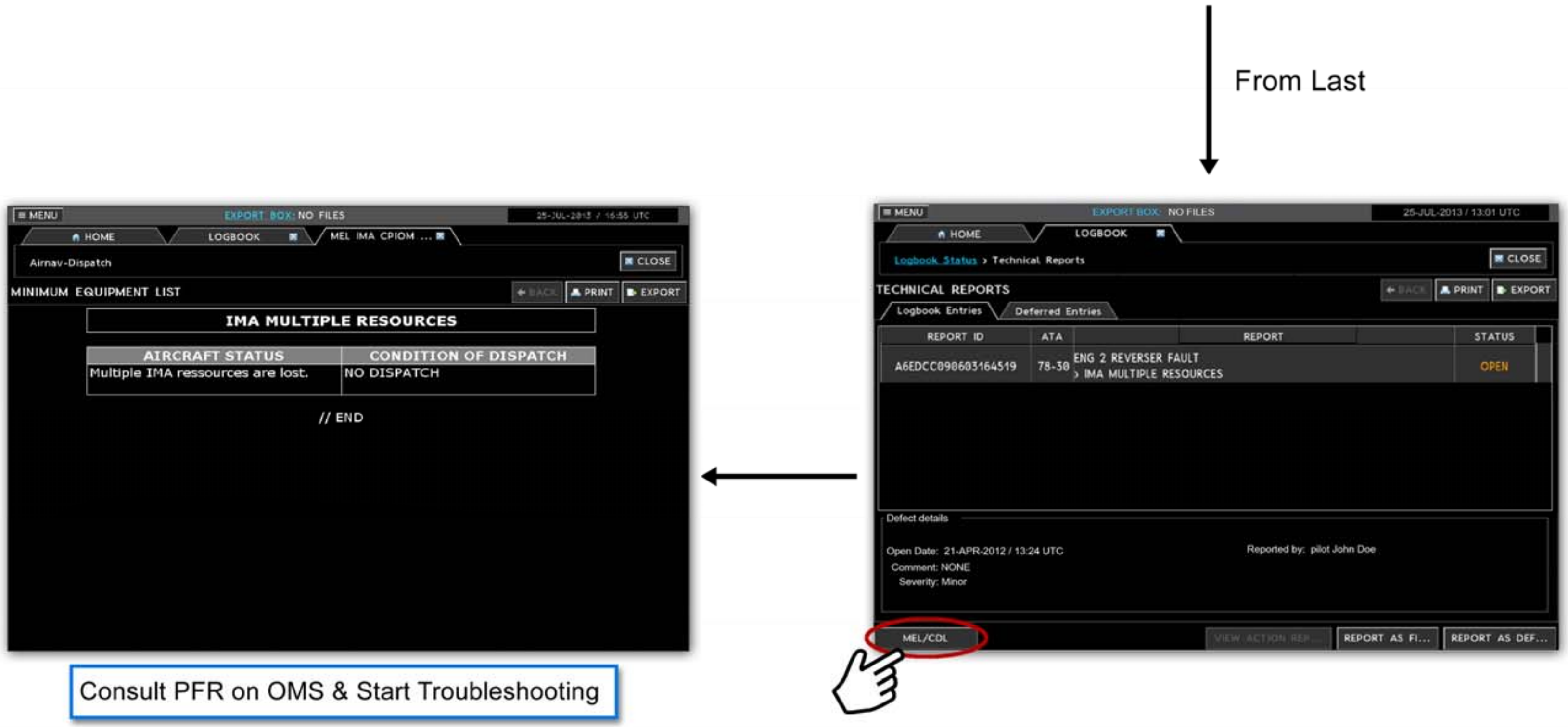
ELECTRONIC LOGBOOK



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ELECTRONIC LOGBOOK

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Aircraft Maintenance Dispatch Workflow Description

Troubleshooting Workflow

OMS home page:

The dispatch condition assessment is done, if the status is NO DISPATCH.

To dispatch the A/C the fault will have to be repaired. To repair the fault, it is necessary to start the troubleshooting phase.

First isolate the fault and identify the root cause. To do this access the PFR function from the OMS.

Post Flight Report (PFR):

View the PFR and select the relevant warning/dispatch message.

Effect Detail Page:

To get access to the related effect detail:

On the effect detail page, select the fault code related to the fault root cause, which hyperlinks to the correlated Aircraft Fault Isolation (AFI) procedure.

Aircraft Fault Isolation (AFI):

In the AFI correlated procedure, all information necessary for the failure isolation and the root cause repair is shown, with hyperlinks to the related maintenance procedures. With good use of the A/C documentation the repair phase can manage from this point.

Release the aircraft:

After completion of the A/C documentation which managed the repair of the fault.

Check the fault is repaired by accessing the ECAM pages and make sure that the warning/dispatch messages have cleared.

After the repair and clean configuration is validated, fill in the logbook (paper or electronic) to answer the flight crew entry.

If there are no other faults, and all the documentation is completed, the maintenance release for the dispatch can be carried out.

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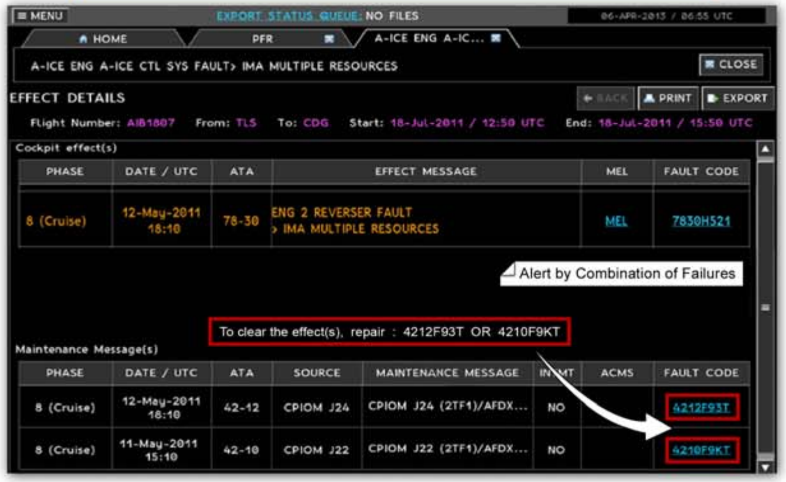
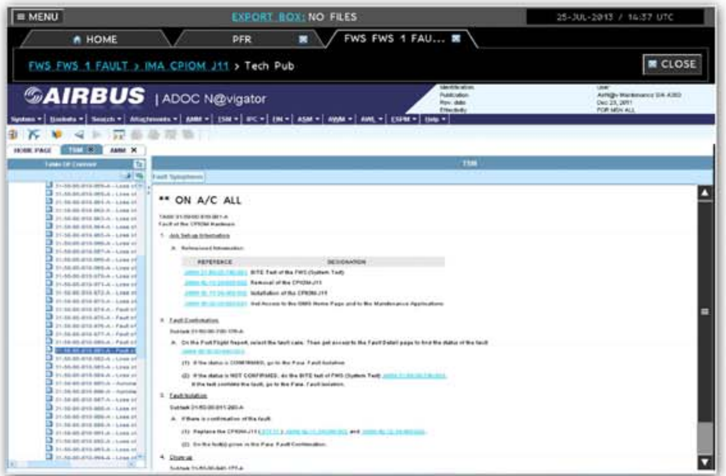
TROUBLESHOOTING WORKFLOW

Access to MEL from OMS PFR Correlated Hyperlink





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TROUBLESHOOTING WORKFLOW

Section 14 – On-board Maintenance System (OMS) & Basic Troubleshooting.

Objectives

The student is now able to:

Identify the components, their purpose, location and interfaces of the On-board Maintenance System (OMS) & Basic Troubleshooting.

- Describe the purpose and operation On-board Maintenance System (OMS).
- Describe how the Trent XWB On-board Maintenance System (OMS) interfaces with A350 aircraft.
- Describe the unique fault code event identifier used on the A350 aircraft.
- Describe using the OMS the Basic Trouble Shooting Work Flow process of the A350 aircraft.

End of Troubleshooting Section

Section 15 – Engine Separation for Air Transportation

Section 15 – Engine Separation for Air Transportation

Aim

To graduate engineering personnel with the skills and knowledge necessary to enable them to perform engine maintenance, functional checks and fault diagnosis to aircraft maintenance manual level, on the Trent XWB engine.

Objectives

At the end of this section the student will be able to:

- State the purpose for Trent XWB engine separation for Air transportation.
- Locate and identify the Fan case and Core stands.
- Separate the Fancase and Core engine.
- Position Fancase and Core engine in their respective transportation positions.
- Identify the storage container and its uses.

Note:

The content of this training manual covers the engine split procedure for the Trent XWB not the engine removal procedure.

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Purpose

The production of the Trent XWB engine has marked the point at which for the first time a large Rolls-Royce commercial turbofan could not always be transported whole using some of the high volume civil air transport options (e.g. Boeing 747-400F and 777-200ERF).

Description

The engine can still be transported complete by other means (e.g. road, sea) and can be transported whole by purpose-built large transport aircraft such as the Antonov 124, 225.

A number of alternative concepts have been evaluated for breaking the engine down into smaller packages that will fit more readily into the hold of cargo aircraft.

From the options evaluated, the most viable is removal of the core section from the fan casing, a similar strategy to that adopted on competitor engines such as the GE90, GEnx, GP7200 and PW4000.

Relative to the Trent XWB the competitor designs offer some advantages from core-mounted accessories, which do not require HP oil and fuel services to be broken.

Operation

The Split Engine Stand (SES) comprises of two separate stands a Fan Case Stand and a Core Stand.

These stands can operate independently of each other and act as storage or transportation tools, but they can be joined together to make one stand for separating or joining an engine.

Each stand has its own method of operation.

Core Stand

The core stand has two mechanical screw jack systems;

- One system enables the stand cradle to be raised and lowered.
- The other system allows the cradle to be moved fore and aft.

Fan Case Stand

The Fan Case Stand has two independent hydraulic systems;

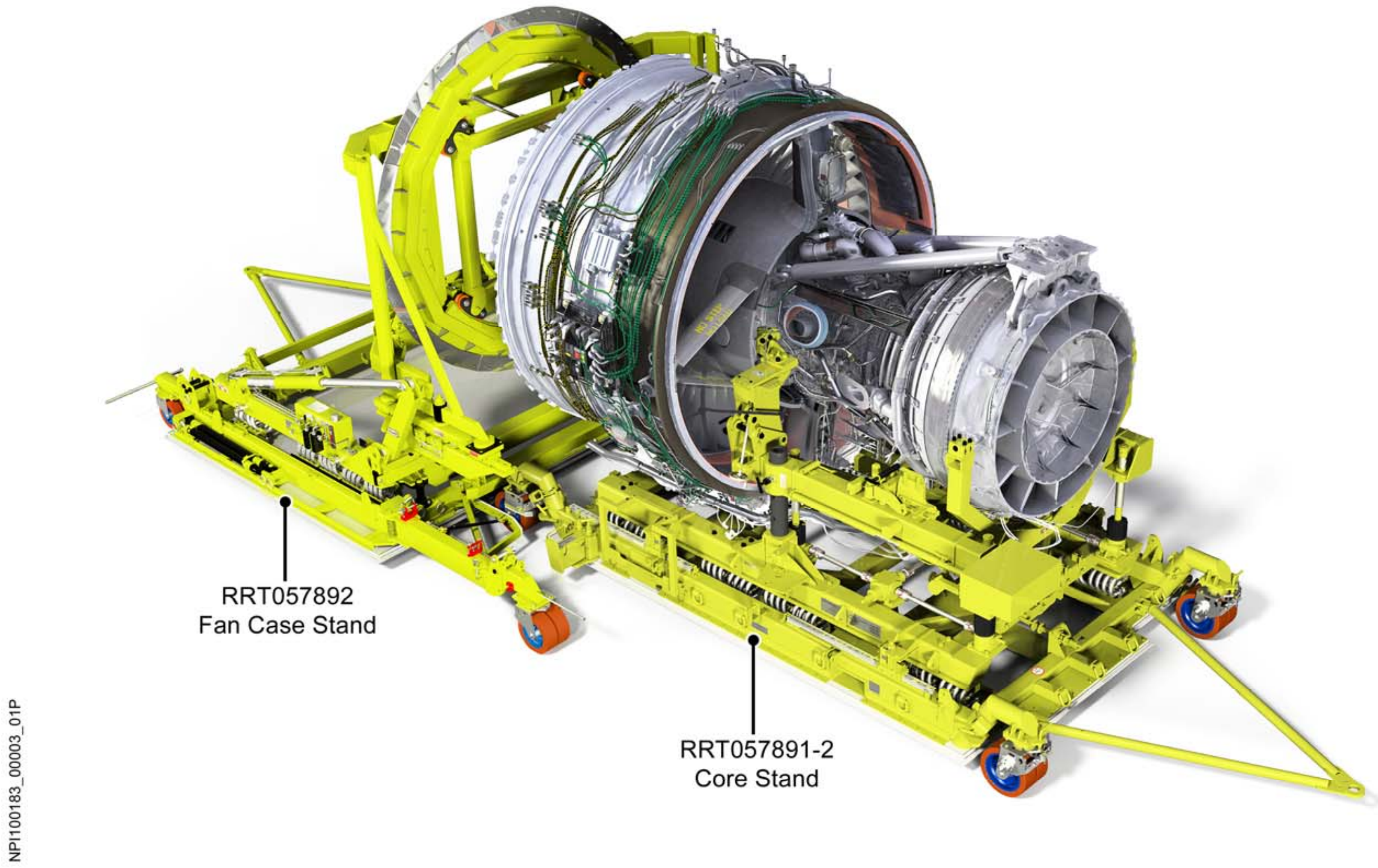
- One system enables the fan case cradle to be raised and lowered.
- The other system allows the cradle to be moved left and right, yaw and pivot around its axis.

Transportation Container

A transportation container is also used as part of the tooling required for engine separation procedure;

- This is to provide a safe and secure storage area for the fan blades, annulus fillers, the spinner and other removed items e.g. tubes, hoses and brackets taken from the separated engine.

A more detailed description and operation of these items are described later in this chapter.



SPLIT ENGINE STANDS

Fan case Stand - RRT057892

Purpose

The purpose of the Fan Case Stand is to enable the fan case to be removed, and refitted to the core engine. It also acts as a transportation vehicle when the fan and core are separated for transit.

Description

When in use the stand stores the non-modular fan case for transportation. It also acts as a tool for removal and refit of the fan case when connected to the core stand, this enables the engine to be separated or assembled together without the use of any external lifting equipment.

Operation

The fan case stand is constructed on a standard double aircraft pallet, and has a fan case frame cradle that has the capability of being raised and lowered around a hinge point on the edge of the fan case stand.

Lowered Position

The fan case frame cradle in the lowered transportation position can move around its central pivot point left and right approximately six inches (15 cm) also it can rotate approximately 48°. This enables the gear box while fitted to the fan case to remain within the profile of the air freight pallet.

Raised Position

The fan case frame cradle in the raised position enables the fan case to be fitted or removed from the core engine. This is achieved by the fan case being bolted to the fan case frame; so the fan can then be securely separated from the core and then lowered ready for transportation.

Stand actuation

The Fan Case Stand movable surfaces are powered by a self-contained hydraulic system, this system has fine (LP) and course (HP) selectors for fast and slow movement. This allows the operator to raise and lower the fan cradle, rotate the cradle 48°, pivot the cradle left and right, and move the cradle in and out from the core engine when in the raised position.

FAN STAND LEADING PARTICULARS	
STATE	WEIGHT (KG)
LOADED	9,200
EMPTY	7,000
WORKING LOAD LIMIT (WLL)	2,200
FANSTAND DIMENTIONS	
STATE STAND ONLY	**UNITS (MM)**
LENGTH	4978
WIDTH	3415
HEIGHT	2150
STATE WITH ENGINE (FAN)	**UNITS (MM)**
LENGTH	4978
WIDTH	3415
HEIGHT	2943

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FANCASE STAND RRT057892

Core Engine Stand - RRT057891-2

Purpose

The purpose of the core stand is to enable the core engine to be removed, and refitted to the Fan Case. It also acts as a transportation vehicle when the fan and core are separated for transit.

Description

When in use the stand stores the core engine for transportation. It also acts as a tool for removal and refit of the core engine when connected to the fan case stand, this enables the engine to be separated or assembled together without the use of any external lifting equipment.

Operation

The core engine stand is constructed on a standard double aircraft pallet, and has a core engine cradle that has the capability of being raised and lowered also moves fore and aft to line up with the fan case stand for engine separation and assembly.

Core Engine Stand Height Positions

The core stand has four height positions these are;

- Split
- Road / on-site bootstrap (engine)
- Road / on-site (core)
- Air (core)

Stand actuation

Raising and lowering (Z Axis). The core engine stand is raised and lowered by a screw jack system which has drive actuated inputs on the left and right hand sides; either one or both will drive the links to actuate the screw jacks.

Fore and aft (X Axis). The core engine stand is moved to or from the fan case stand by a screw jack system that is actuated by a drive that is located at the rear of the stand. This will move the core engine support system only via a sledge type core cradle and will not move the stand on its support wheels.

CORE STAND LEADING PARTICULARS	
CORE STAND WEIGHT	
STATE	**WEIGHT (KG)**
LOADED	17,500
EMPTY	8,500
WORKING LOAD LIMIT (WLL)	9,000
CORE STAND DIMENTIONS	
STATE STAND ONLY	**UNITS (MM)**
LENGTH	5270.5
WIDTH	2986.4
HEIGHT	2450
STATE WITH ENGINE (CORE)	**UNITS (MM)**
LENGTH	5270.5
WIDTH	3605
HEIGHT (TO FWD ENGINE MOUNT)	4035

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CORE STAND RRT057891-2

Transportation box - RRT070226

Purpose

The purpose of the transportation box is to store and transport safely the removed items from the engine separation process.

Description

The transportation box is built around a standard aircraft pallet footprint and is divided into three sections.

- An upper section.
- A lower section.
- A covering lid.

Operation

The lower section stores removed items from the bifurcation area of the engine, upper and lower splitter fairings, the gas generator fairings and other miscellanies items. This has a lid that divides and protects the lower box contents from the base of the upper section.

The upper section stores the fan blades, the annulus fillers, and the main engine hydraulic lines. This has a covering box type lid that makes the container safe and ridged for transportation and stowage purposes. It holds a four point lifting sling for lifting and lowering of the transportation box upper and lower lids.
This four point sling is stored in a purpose built in cabinet within the upper lid when not in use.

The Lid covers the upper section and forms a protective

TRANSPORTATION BOX RRT070226	
STATE	WEIGHT (KG)
LOADED	4,600
EMPTY	4,057
DIMENTIONS	
STATE	**UNITS (MM)**
LENGTH	2322
WIDTH	3062
HEIGHT	2347

enclosure for the ban blades, annulus fillers and hoses.

Upper Level

Lower Level

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TRANSPORTATION BOX RRT070226

Section 15 – Engine Separation for Air Transportation

Objectives

At the end of this section the student will be able to:

At the end of this section the student will be able to:

- State the purpose for Trent XWB engine separation for Air transportation.
- Locate and identify the Fan case and Core stands.
- Separate the Fancase and Core engine.
- Position Fancase and Core engine in their respective transportation positions.
- Identify the storage container and its uses.

Note

The content of this training manual covers the engine split procedure for the Trent XWB not the engine removal procedure.

End of the Trent XWB Engine Separation Section